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A SIMPLIFIED ANALYSIS OF PROPULSION INSTALLATION LOSSES FOR COMPUTERIZED AIRCRAFT DESIGN

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16. Abstract

A simplified method is presented for computing the installation losses of aircraft gas-turbine propulsion systems. The method has been programmed for use in computer-aided conceptual aircraft design studies that cover a broad range of Mach numbers and altitudes. The items computed are: inlet size, pressure recovery, additive drag, subsonic spillage drag, bleed and bypass drags, auxiliary air systems drag, boundary-layer diverter drag, nozzle boattail drag, and the interference drag on the region adjacent to multiple nozzle installations. The methods for computing each of these installation effects are described and computer codes for the calculation of these effects are furnished. The results of these methods are compared with selected data for the F-5A and other aircraft. The computer program can be used with uninstalled engine performance information which is currently supplied by a cycle analysis program. The program, including comments, is about 600 FORTRAN statements long, and uses both theoretical and empirical techniques.

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NOTATION

The notation used in the following sections is defined with the corresponding FORTRAN name used in the program indicated parenthetically. Figure 1 shows the nomenclature used for the various inlet and nozzle locations. The values below are defined per engine and the drag coefficients are based on inlet capture area unless noted. The starred (*) items are required program inputs which are either user input or are supplied by another subroutine in the aircraft synthesis program.

Symbol	Code	
A		area, m^2, ft^2
A_{AUX}/A_{ENG}	(AUAENG)*	auxiliary systems area ratio
A_{BL}/A_c	(ABLEAC)	bleed mass flow ratio
A_{BP}/A_c	(ABYPAC)	bypass mass flow ratio
A_c	(AC)	inlet capture area (per engine), m^2, ft^2
A_{CC}	(ACC)	area of exit nozzle (joint point between engine and fuselage)
A_E		area of exit, m^2, ft^2
A_{EF}	(AEF)*	engine face flow area (per engine), m^2, ft^2
A_{ENG}	(AENG)*	engine face total area (per engine), m^2, ft^2
A_{EXIT}	(AEXIT)	nozzle exit area (per engine), m^2, ft^2
$A_{NOZ_{TH}}$	(ANOZT)	nozzle throat area (per engine), m^2, ft^2
A_o	(AO)	area of free-stream stream tube (per engine), m^2, ft^2
A_o/A_c	(AOAC)	mass flow ratio of inlet (per engine), m^2, ft^2
A_s	(AS)	projected frontal area of compression surface, m^2, ft^2
A_{TH}	(AT)	inlet throat area (per engine), m^2, ft^2
A_{TH_D}	(ATD)	inlet throat area (per engine) at M_{DES}, m^2, ft^2
A_{VENT}/A_c	(AVEACD)	ratio of engine ventilation flow area to inlet capture area (per engine)

A_{WEDGE}/A_c	(AWAENG)*	boundary-layer diverter area ratio
A_y	(AY)	projected frontal area of compression surface forward of point of normal shock impingement, m^2, ft^2
C_D		drag coefficient
$C_{D_{AD}}$	(CDAD)	supersonic spill additive drag coefficient
$C_{D_{AUX}}$	(CDAUX)	auxiliary systems drag coefficient
$C_{D_{BL}}$	(CDBE)	bleed drag coefficient
$C_{D_{BP}}$	(CDBP)	bypass drag coefficient
$C_{D_{BT}}$	(CDBT)	nozzle boattail drag coefficient
$C_{D_{DIV}}$	(CDDIV)	boundary-layer diverter drag coefficient
$C_{D_{INF}}$	(CDI)	nozzle interference drag coefficient
C_{D_β}		boattail drag coefficient based on A_{CC}
$C_{P_{DIV}}$		pressure coefficient on diverter surface
C_{P_S}	(CPCS)	pressure coefficient on compression surface
C_S or $C_{D_{SP}}$	(CS or CDADS)	subsonic spill additive drag coefficient
C_T		thrust coefficient
D_{CC}	(DCC)	nozzle diameter at customer connect, m, ft
D_{ENG}	(DENG)	engine face diameter, m, ft
g		acceleration of gravity, m/sec^2, ft/sec^2
D_g	(DEXIT)	nozzle exit diameter, m, ft
h		altitude, m, ft
IPR	(IPR)*	inlet pressure recovery code
L		distance between normal shock position and inlet lip

L/y_c	(XLVD)	distance between normal shock position and inlet lip ratioed to inlet capture diameter
L_{NOZ}	(XLNOZ)	nozzle length, m, ft
M		Mach number
$\dot{m}$		mass flow, kg/sec, lb/sec
$\dot{m}_{AUX}$		auxiliary systems mass flow, kg/sec, lb/sec
$\dot{m}_{BP}$		bypass mass flow, kg/sec, lb/sec
M_{cone}	(XMCONE)	compression surface Mach number
M_{DES}	(XMDES)*	inlet design Mach number
M_E		exit Mach number
$\dot{m}_E$		exit mass flow, kg/sec, lb/sec
M_{EF}	(XMEF)*	engine face Mach number
M_{EXIT}	(XMEX)	nozzle exit Mach number
M_{TH}	(XMT)*	inlet throat Mach number
M_∞	(XMO)*	free-stream Mach number
N_{ENG}	(EN)*	number of engines
NPR	(NPR)*	nozzle pressure ratio
P		static pressure, N/m^2, lb/ft^2
P_E		exit static pressure, N/m^2, lb/ft^2
PR_{DES}	(PRDES)	supersonic diffuser pressure recovery at M_{DES}
PR_{SUB}	(PRSUB)	subsonic diffuser pressure recovery
PR_{SUP}	(PR)	supersonic diffuser pressure recovery
PR_{TOT}	(PRTOT)	total pressure recovery to engine face
PSPIN		cone surface pressure ratio
P_t		total pressure, N/m^2, lb/ft^2
$P_{t_{E_{Bleed}}}$	(PTBLE)	bleed exit total pressure, N/m^2, lb/ft^2

$P_{t_{E_{Bypass}}}$	(PTBYP)	bypass exit total pressure, N/m^2, lb/ft^2
$P_{t_{EF}}$		total pressure at engine face, N/m^2, lb/ft^2
P_{TH}		cone static pressure at the throat, N/m^2, lb/ft^2
$P_{t_{NOZ}}$	(PTNOZ)*	nozzle exit total pressure, N/m^2, lb/ft^2
$P_{t_{TH}}$		total pressure at inlet face, N/m^2, lb/ft^2
P_{t_∞}	(PTO)*	free-stream total pressure, N/m^2, lb/ft^2
P_∞	(PINF)*	free-stream static pressure, N/m^2, lb/ft^2
Q or q_∞	(Q)*	free-stream dynamic pressure, N/m^2, lb/ft^2
SFC		specific fuel consumption, kg/N-hr, lb/lb-hr
S/D_g	(SODG)*	nozzle spacing ratio
S_{ref}	(SWING)*	wing reference area, m^2, ft^2
T		thrust, N, lb
T_g	(FIP)*	gross thrust per engine, N, lb
T_t		total temperature, K, R
$T_{t_{NOZ}}$	(TTNOZ)*	nozzle exit total temperature, K, R
T_{t_∞}	(TTO)*	free-stream total temperature, K, R
V_E		exit velocity, m/sec, ft/sec
V_∞		free-stream velocity, m/sec, ft/sec
W_a	(WA)*	engine airflow, kg/sec, lb/sec
X_{cone}/yc	(XCOYC)	distance from cone tip to inlet face ratioed to inlet capture diameter
y_c	(YC)	inlet capture diameter, m, ft
y_s		diameter of inlet centerbody at inlet throat, m, ft
β	(BETA)	nozzle boattail angle, deg

ΔPR	(DELPR)*	incremental pressure recovery correction
γ	(GAMMA)	isentropic constant
λ	(LAMBDA)	angle at inlet lip between average direction of flow and longitudinal axis of inlet
ρ_∞	(RHO)	free-stream static density, kg/m^3, lb/ft^3
θ	(THETA)	cone half angle, deg
θ_D	(THDIV)	boundary-layer diverter wedge angle, deg
θ_E		exit angle, deg (COSDE is cosine of exit angle in program)

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SUMMARY

A simplified method is presented for computing the installation losses of aircraft gas-turbine propulsion systems. The method has been programmed for use in computer-aided conceptual aircraft design studies that cover a broad range of Mach numbers and altitudes. The items computed are: inlet size, pressure recovery, additive drag, subsonic spillage drag, bleed and bypass drags, auxiliary air systems drag, boundary-layer diverter drag, nozzle boat-tail drag, and the interference drag on the region adjacent to multiple nozzle installations. The methods for computing each of these installation effects are described and computer codes for the calculation of these effects are furnished. The results of these methods are compared with selected data for the F-5A and other aircraft. The computer program can be used with uninstalled engine performance information which is currently supplied by a cycle analysis program. The program, including comments, is about 600 FORTRAN statements long, and uses both theoretical and empirical techniques.

INTRODUCTION

The design of advanced aircraft systems requires the consideration of many different tradeoffs and parameters to arrive at an optimum design for a particular requirement or group of requirements. One is the effect of interaction between the aerodynamics and the propulsion of these systems. Propulsion installation effects on high-speed aircraft can amount to 10 percent or more of the aircraft drag and can also degrade the propulsion thrust via inlet total-pressure recovery penalties and nozzle-flow penalties. These effects are significant in high-speed aircraft design, and thus require attention, even in early design studies.

Tradeoff studies are usually done manually or, more recently, by many large computer programs with manual communication between them. As computer capabilities have increased, it has become possible to communicate between these disciplines within the computer in an automated or integrated fashion. This integration allows computation of the trajectory of the aircraft over its entire mission, thereby providing the ability to determine the effects of various parameters and to optimize the aircraft for specific requirements subject to various constraints. The method and computer code presented in this report is intended to supply the propulsion installation losses as required in this process. The code is designed to work as part of a propulsion module

in the framework of the Aircraft Synthesis Program, ACSYNT (fig. 2), which has been developed at the Ames Research Center (ref. 1).

The purpose of this report is to document the methods and the computer code for propulsion installation losses as presently employed in ACSYNT. Limited example comparisons of calculations with data are made and areas of further research identified. It should be emphasized that, at present, the methods are preliminary in nature and further work is needed to improve the techniques and to perform additional correlations with data.

PROGRAM PHILOSOPHY

The purpose of the *Pr*opulsion *In*stallation *C*alculation (PRINC) module is to compute the air induction system and nozzle/afterbody effects in the ACSYNT program. The procedures employed in the present subroutine are general, since the methods must be applicable to a variety of inlet, engine, and nozzle types over a broad range of Mach numbers and altitudes. An additional important requirement is that the calculations be very rapid, since installation losses are computed many times (over 1000) in a run of the ACSYNT program.

Figure 3 shows a block diagram of the method. A modular approach is used so that future additions and improvements can be easily incorporated. Items computed include (1) inlet pressure recovery, (2) inlet size, (3) additive and spillage drags, (4) bleed and bypass drags, (5) auxiliary system drag, (6) boundary-layer diverter drag, (7) nozzle boattail drag, and (8) nozzle interference drag. In figure 3, those parameters listed inside the boxes are output from the various modules and those parameters listed beneath each box are required inputs to each module.

There are varied accounting approaches for the aerodynamic propulsion system and propulsion system/airframe interaction losses. The method employed in the PRINC module is to charge all losses (listed above) to the engine thrust and specific fuel consumption (SFC) as indicated in figure 4. However, the total propulsion installation drag as well as the individual propulsion-related drags are computed separately so that any desired accounting method may be adopted by the user. An available option in the program is a multiplying factor for any or all of the propulsion installation losses to adjust the level of these penalties at the user's discretion.

DESCRIPTION OF METHODS

This section documents the methods used in the propulsion installation loss module (PRINC) and diagrammed in figure 3. It is assumed, for the inlet drag calculations, that the inlet is an axisymmetric, external compression design and, for the additive drag calculation, that the surface pressures are for a cone of an average half angle of 20°. The drag coefficients computed in the following development are based on inlet capture area, except where

noted. The equations, derivations, and programming details are presented in appendix A. A FORTRAN listing of all the modules is included in appendix B.

Inlet Pressure Recovery — The inlet pressure recovery is divided into two parts, the pressure recovery in the region ahead of the inlet face and the pressure recovery in the subsonic diffuser after the inlet face. The pressure recovery in the region ahead of the inlet face is estimated by the use of the standard AIA or Military Specification 5008B methods or by the assumption of normal shock pressure recovery (appendix A). The pressure recovery versus Mach number computed by these three methods is shown in figure 5.

The subsonic diffuser pressure recovery is estimated by the empirical method of Ball (ref. 2), which gives this pressure recovery as a function of the throat Mach number, the inlet lip bluntness, and the free-stream Mach number. For the present study, the inlet lip has been assumed to be sharp and, thus, the inlet subsonic diffuser pressure recovery is independent of lip bluntness or free-stream Mach number. Also, the geometric inlet throat Mach number is equal to the effective inlet throat Mach number as described in reference 2.

A fourth method available in the program is to input the inlet total pressure recovery as a function of free-stream Mach number in tabularized form.

Inlet Sizing — The inlet face flow area is determined by a mass balance (conservation of mass) between the inlet face and the engine face. The mass flow at the engine face is determined by the requirements of the engine. The inlet face flow area is increased over that of the engine to allow for bypass, bleed, and powerplant ventilation mass-flow requirements. The free-stream stream-tube cross-sectional area is determined by a mass balance between the free stream and the inlet face. The inlet design Mach number is used to define the inlet capture area, which is equal to the free-stream stream-tube cross-sectional area at the engine's maximum power setting. The inlet capture area is held constant at off-design conditions; however, the centerbody is allowed to move so that the inlet throat Mach number is held at some specified value. No check is made on the mechanical difficulty of achieving this variation. The key assumption in this analysis is that the inlet throat Mach number is constant. The programming details of this subroutine are included in appendix A.

Additive Drag — The engine thrust is referenced to free-stream conditions. The loss in momentum of the airflow ahead of the inlet system must be accounted for in the bookkeeping system. This loss in momentum ahead of the inlet face is called "additive drag" and is a function of the inlet geometry, the free-stream Mach number, and the mass flow of the engine.

The inlet additive drag is computed by a momentum balance between the inlet face and the free stream. The cosine of average flow angle (with respect to the inlet centerline) at the inlet face is assumed to be 1.0. The inlet is assumed to be external compression (that is the normal shock is outside of the cowl lip). The inlet throat Mach number is held constant at some specified

value. The inlet geometry is assumed to be axisymmetric. The additive drag can be computed (ref. 3) from

$$C_{D_{AD}} = \frac{2}{\gamma M_\infty^2}\left[\frac{A_{TH}}{A_c}\frac{P_{t_\infty}}{P_\infty}\frac{P_{t_{TH}}}{P_{t_\infty}}\frac{P_{TH}}{P_{t_{TH}}}(\gamma M_{TH}^2 + 1)\cos\lambda + \frac{A_c - A_{TH}}{A_c}\frac{\overline{P}_{cone}}{P_\infty} - 1.0 - \frac{A_o}{A_c}\gamma M_\infty^2\right] + C_S$$

The cone pressure calculation uses a polynomial approximation presented by Lighthill (ref. 4). The subsonic spillage effect C_S is computed using an empirical technique described by Sibulkin (ref. 3). A complete description of the method is included in appendix A.

Bypass Drag — In high-Mach-number aircraft design the inlet is usually sized at the maximum design Mach number. During off-design operation at lower Mach numbers, the inlet usually has the capacity to supply an excess airflow to the engine. This excess airflow must be either taken onboard the aircraft and passed (bypassed) around the engine or diverted (spilled) around the inlet system.

The bypass drag is computed from a momentum balance between the free-stream and the bypass exit. The bypass exit nozzle can be either sonic or fully expanded. After considerable simplification (see appendix A), the momentum balance yields

$$\frac{C_D}{(A_{BP}/A_c)} = 2\left[1 - \cos\theta_E \frac{M_E}{M_\infty}\left(\frac{1 + 0.2M_\infty^2}{1 + 0.2M_E^2}\right)^{0.5}\right] + \left\{\frac{\cos\theta_E}{0.7M_\infty^2}\frac{M_\infty}{M_E}\left(\frac{1 + 0.2M_E^2}{1 + 0.2M_\infty^2}\right)^3\left[\frac{1}{(P_{t_E}/P_{t_\infty})} - \left(\frac{1 + 0.2M_\infty^2}{1 + 0.2M_E^2}\right)^{3.5}\right]\right\}$$

where γ is assumed to be 1.4. If it is assumed that the bypass exit nozzle is sonic, then

$$M_E = 1.0$$

If it is assumed that the bypass exit nozzle is fully expanded, then

$$P_E = P_\infty$$

$$M_E = [5(P_{t_E}/P_\infty)^{0.286} - 1)]^{0.5}$$

The bypass exit pressure recovery is assumed to be a fraction of the inlet total pressure recovery (to the engine face). Typical values for this fraction are

$$P_{t_E}/P_{t_\infty} = KP_{t_{EF}}/P_{t_\infty}$$

where $0.3 \leq K \leq 0.7$.

Bleed Drag — The inlet compression ramp or cone for typical supersonic inlet designs often have a considerable length exposed to an adverse pressure gradient. This can create a boundary layer which is thick enough to cause losses in engine performance. The problem is particularly acute in regions where a shock wave interacts with this boundary layer. In order to maintain efficient engine performance, part of the boundary layer is removed on these compression surfaces in some inlets, and it is necessary to account for the momentum loss of this bleed flow. A momentum balance between the free stream and the bleed exit yields an expression similar to the bypass drag formulation. The bleed exit can be assumed to be either sonic or fully expanded. The momentum balance yields

$$\frac{C_D}{(A_{BL}/A_c)} = 2\left[1 - \cos\theta_E \frac{M_E}{M_\infty}\left(\frac{1 + 0.2M_\infty^2}{1 + 0.2M_E{}^2}\right)^{0.5}\right]$$
$$+ \left\{\frac{\cos\theta_E}{0.7M_\infty^2}\frac{M_\infty}{M_E}\left(\frac{1 + 0.2M_E{}^2}{1 + 0.2M_\infty^2}\right)^3\left[\frac{1}{(P_{t_E}/P_{t_\infty})} - \left(\frac{1 + 0.2M_\infty^2}{1 + 0.2M_E{}^2}\right)^{3.5}\right]\right\}$$

where γ is assumed to be equal to 1.4. If it is assumed that the bleed exit nozzle is sonic, then

$$M_E = 1.0$$

If it is assumed that the bleed exit nozzle is fully expanded, then

$$P_E = P_\infty$$

$$M_E = [5(P_{t_E}/P_\infty)^{0.286} - 1]^{0.5}$$

The bleed exit pressure recovery is assumed to be a fraction of the inlet total pressure recovery (to the engine face). Typical values for this fraction are

$$P_{t_E}/P_{t_\infty} = KP_{t_{EF}}/P_{t_\infty}$$

where $0.3 \leq K \leq 0.7$.

A complete derivation of these equations is contained in appendix A.

Auxiliary Systems Drag — The auxiliary systems drag accounts for the airflow taken into the aircraft for systems cooling and auxiliary power generation. Many aircraft have small auxiliary inlets mounted at some convenient place to serve this purpose, and the drag created can be significant. It is assumed that the total momentum of the flow into these systems is lost. Therefore the auxiliary system drag is

$$C_{D_{AUX}} = \frac{\dot{m}_{AUX} V_\infty}{Q A_c} = \frac{\rho_\infty A_{AUX} V_\infty^2}{\frac{1}{2} \rho_\infty V_\infty^2 A_c} = 2 \frac{A_{AUX}}{A_c}$$

where A_{AUX}/A_c is the ratio of the auxiliary system inlet capture area to aircraft inlet capture area. Typical values for this quantity range from 0.005 to 0.01.

Boundary-Layer Diverter Drag — In many inlet installation systems, the inlets are located close to the aircraft's larger components (i.e., wings, fuselage) which generate regions of low momentum ahead of the inlet. The ingestion of these boundary layers into the inlet creates a nonuniform flow distribution which can cause considerable performance degradation in the engine. This problem has been avoided by the addition of a ramp (a plow) between the inlet and the boundary-layer generating surface. The turning of the flow in these systems adds drag to the aircraft, which must be accounted for. A fit of data (refs. 5 and 6) yields

$$C_{D_{DIV}} = \frac{1.2}{M_\infty^2} \frac{\theta_D}{20} \frac{A_{WEDGE}}{A_c} \quad ; \quad M_\infty \geq 1.55$$

$$= 0.499 \frac{\theta_D}{20} \frac{A_{WEDGE}}{A_c} \quad ; \quad 0.95 \leq M_\infty \leq 1.55$$

$$= 0.499 \frac{M_\infty - 0.8}{(0.95 - 0.80)} \frac{\theta_D}{20} \frac{A_{WEDGE}}{A_c} \quad ; \quad 0.80 \leq M_\infty \leq 0.95$$

$$= 0.0 \quad ; \quad M \leq 0.8$$

Details on the data and a comparison with the fit are given in appendix A.

Boattail Drag — The boattail drag on the airframe back to the point where the nacelle and engine are joined (see fig. 1b) is calculated as part of the aircraft drag. The boattail drag on the portion of the engine which includes the engine nozzle after this joint is charged to the engine performance in the present accounting system. The boattail drag estimation method used is an empirical technique developed by Ball (ref. 2) from wind-tunnel data on isolated boattail nozzles. The nozzle interference drag described in the next section corrects this for installations of more than one engine. The boattail drag is based on the area at the point where the engine is joined to the airframe. The formulation is for an engine nozzle pressure ratio (engine exit

total pressure to free-stream static pressure) of 2.5; however, correction terms are included for different nozzle pressure ratios. The engine nozzle exit area is computed from the engine thermodynamic data. The boattail angle is computed from the engine diameter and the assumption that the length of the boattail is equal to the engine diameter. It is also assumed that the diameter of the boattail at the connection point between the engine and aft fuselage or nacelle is 10 percent greater than the engine diameter. A complete description of this procedure is included in appendix A.

Nozzle Interference Drag — The nozzle interference drag accounts for the drag on the base area between multiple nozzles. The independent variables are free-stream Mach number and nozzle spacing ratio S/D_g (ratio of the distance between nozzle centerlines to nozzle exit diameter). The calculation technique, developed by Ball (ref. 2) from wind-tunnel data, estimates the ratio of the drag due to nozzle interference divided by ideal gross thrust at a nozzle pressure ratio of 2.5. This value is corrected to a drag coefficient based on inlet capture area. A complete description of this computation is included in appendix A.

EXAMPLE CALCULATIONS

This section presents example computations from the PRINC module of typical installation drags, net propulsive thrust, and specific fuel consumption values. After PRINC module calculations of inlet mass flow and propulsion installation drags for a simulated F-5A are presented, these results are then used to determine the overall installed thrust and SFC of an ACSYNT simulated F-5A. Comparisons are made of these results with F-5A flight test data.

Mass Flow Summary

The effect of Mach number on engine mass flow ratio A_o/A_c for the PRINC module simulated F-5A is presented in figure 6. Note that the F-5A has no bleed or bypass. The spillage mass flow is the difference between $A_o/A_c = 1.0$ and the A_o/A_c set by the engine (plotted). This difference would be much larger for an aircraft with a higher inlet design Mach number M_{DES}. The method is capable of handling bleed and bypass in the manner described in the section on bypass and bleed drag.

Total Installation Drag

Figure 7 is an example PRINC module calculation of the installation drag coefficients based on wing reference area as a function of M_∞ for a simulated F-5A inlet system. The total installation drag coefficient is shown, as well as the various components for maximum afterburning (A/B) and military power settings. For this same inlet system, the effects of engine throttling at $M = 0.9$ and 1.2 are shown in figures 8a and b.

Net Propulsive Thrust Correlation

A comparison of the thrust calculated by the ACSYNT propulsion subroutine and the PRINC module with data determined from F-5A flight tests is shown in figures 9a and b for maximum A/B and military power settings. The results are presented for two engines over a range of Mach numbers at 10 973 m (36 000 ft). The upper portion of each figure compares the uninstalled thrust from the ACSYNT propulsion module with corresponding values from the J-85-GE-13 engine specifications (ref. 7). Both thrust values are based on the AIA standard ram recovery schedule. The table shows the percentage difference between the calculated results and data for selected Mach numbers; that is,

$$\frac{\text{Calculated-Actual}}{\text{Actual}} \times 100$$

The lower portion of the figure shows a comparison between the installed thrust calculated by the ACSYNT propulsion subroutine with corrections calculated by the PRINC module and flight-test modified data from reference 8. The PRINC module calculations include corrections for a pressure recovery schedule based on a corrected airflow of 20.4 kg/sec (45 lb/sec) (ref. 9) and for the following installation losses — additive drag, auxiliary systems drag, boundary-layer diverter drag, and nozzle boattail and interference drags. Bleed and bypass drags are zero. Exactly what corrections are included in the flight-test modified data of reference 8 is not clear, but it is suspected that losses for the boundary-layer diverter and the nozzle are not included. This would account for some of the overcorrection by the PRINC module. With a few exceptions, the percentage differences for both power settings are within 10 percent.

SFC Correlation

Figures 10a and b show comparisons between specific fuel consumption values from the ACSYNT propulsion subroutine and the PRINC module and data determined from F-5A flight tests. These comparisons correspond to the thrust correlations shown in figures 9a and b. As with thrust, the percentage differences are generally within 10 percent. It should be noted that the F-5A flight-test evaluation may use a different method of bookkeeping, which could account for some of the differences.

CONCLUDING REMARKS

A simplified method has been presented for computing the installation losses of aircraft gas-turbine propulsion systems. The program employs rapid and sufficiently accurate estimating procedures suitable for use in computer-aided conceptual design studies of aircraft systems over a broad range of Mach numbers and altitudes. The items which can be computed are: inlet size and pressure recovery, additive drag, subsonic spillage drag, bleed and bypass drag, auxiliary air systems drag, boundary-layer diverter drag, nozzle boattail

drag, and the interference drag on the region adjacent to multiple nozzle installations. The methods for computing each of these installation effects have been described and compared with either data or the results of more elaborate computing procedures. Finally, a comparison of the overall results of the method with F-5A performance specifications indicates an accuracy within about 10 percent in installed thrust and specific fuel consumption. This is considered sufficiently accurate for computerized design at the early stages of vehicle definition.

APPENDIX A

DEVELOPMENT OF PROGRAMMED EQUATIONS

This appendix contains a brief development and description of the equations that are used in the PRINC program. The equations are presented by subroutine.

INLET PRESSURE RECOVERY

(MODULES PRSUBS AND PRINL)

This section is divided into two modules, one to calculate the subsonic diffuser pressure recovery PR_{SUB} and another to calculate both the supersonic diffuser recovery PR_{SUP} and the total pressure recovery to the engine face PR_{TOT}.

Subsonic Diffuser Recovery

The empirical method of reference 2 is used. For $\gamma = 1.4$,

$$PR_{SUB} = \frac{P_{t_{EF}}}{P_{t_{TH}}} = 1.0 - EPS\left\{1.0 - \frac{1.0}{[1.0 + 0.2(M_{TH})^2]^{3.5}}\right\}$$

where

$$EPS = 0.37148(M_{TH})^2 - 0.231428(M_{TH}) + 0.06$$

Supersonic Diffuser Recovery

Four different options are available for calculating the supersonic diffuser recovery:

(1) AIA standard ram recovery — From reference 10, we have

$$PR_{SUP} = \frac{P_{t_{TH}}}{P_{t_\infty}} = 1.0 \quad ; \qquad M_\infty \leq 1.0$$

$$PR_{SUP} = \frac{P_{t_{TH}}}{P_{t_\infty}} = 1.0 - 0.1(M_\infty - 1.0)^{1.5} \quad ; \qquad M_\infty > 1.0$$

(2) Military Specification 5008B — Also from reference 10, we have

$$PR_{SUP} = \frac{P_{t_{TH}}}{P_{t_\infty}} = 1.0 \; ; \qquad M_\infty \leq 1.0$$

$$PR_{SUP} = \frac{P_{t_{TH}}}{P_{t_\infty}} = 1.0 - 0.075(M_\infty - 1.0)^{1.35} \; ; \qquad M_\infty > 1.0$$

(3) Normal shock — From reference 11, we have

$$PR_{SUP} = \frac{P_{t_{TH}}}{P_{t_\infty}} = 1.0 \; ; \qquad M_\infty \leq 1.0$$

$$PR_{SUP} = \frac{P_{t_{TH}}}{P_{t_\infty}} = \left(\frac{6M_\infty^2}{M_\infty^2 + 5.0}\right)^{7/2} \left(\frac{6}{7M_\infty^2 - 1.0}\right)^{5/2} \; ; \qquad M_\infty > 1.0$$

(4) Input table of PR_{SUP} vs M_∞ — See program listing in appendix B.

Figure 5 shows a comparison of the first three supersonic diffuser pressure recovery schedules described above.

The particular total pressure recovery schedule to be used is selected by use of the control parameter IPR, as follows:

IPR Code	Recovery schedule
= 1,	AIA standard ram recovery — ΔPR
= 2,	MIL Specification 5008B — ΔPR
= 3,	normal shock — ΔPR
= 4,	table look up

where ΔPR is an input incremental pressure recovery correction. If IPR is positive the installation effects are included. If IPR is input with a minus sign, the installation effects are neglected and the thrust is corrected only for the pressure recovery losses (i.e., IPR = -1 gives AIA ram recovery — ΔPR and no installation losses).

If IPR is input as a positive number, but preceded by a one (i.e., 11, 12, 13, or 14), the installation effects are included and the subsonic diffuser pressure recovery is computed from the empirical results of reference 2 (see subsonic diffuser recovery in the previous section). Thus, IPR = 11 gives the AIA ram recovery multiplied by PR_{SUB} with the installation effects included.

The subsonic diffuser pressure recovery is multiplied by the supersonic diffuser pressure recovery to give the total pressure recovery to the engine face. That is,

$$PR_{TOT} = \frac{P_{t_{EF}}}{P_{t_{TH}}} \times \frac{P_{t_{TH}}}{P_{t_\infty}} = PR_{SUB} \times PR_{SUP}$$

Also in this module, the supersonic diffuser pressure recovery at the inlet design Mach number (PR_{DES} at M_{DES}) is multiplied by the subsonic pressure recovery to give the total pressure recovery to the engine face.

INLET SIZING (MODULE SIZIN)

This module is used to compute the inlet capture area A_c. The inlet capture area is defined to be the total projected frontal area of the inlet, including the projected frontal area of the centerbody (see fig. 1). The inlet capture area is computed at the design Mach number, altitude, and power setting, and is held fixed for off-design operation.

A useful relationship which is needed in the following development is the corrected airflow per unit area, which is defined to be

$$WFF \equiv \frac{W_a\sqrt{T_t}}{P_t A} = g\sqrt{\frac{\gamma}{R}}\, M\left(1 + \frac{\gamma - 1}{2} M^2\right)^{-\left(\frac{\gamma+1}{2(\gamma-1)}\right)}$$

$$= 0.92M\left(\frac{1}{1 + 0.2M^2}\right)^3 \quad ; \qquad \gamma = 1.4, \quad g = 32.2, \quad \text{and} \quad R = 1716$$

WFF(M) denotes the corrected airflow per unit area (sometimes called the weight flow function) calculated for the Mach number specified in the parenthesis. For example, $WFF(M_{EF})$ means the weight flow function calculated for the engine face Mach number.

Inlet Throat Area

For external compression inlet designs with sharp lips the inlet face flow area is equal to the inlet throat area. The inlet throat Mach number is input to the program and the engine face Mach number and engine face flow area A_{EF} are obtained from the engine description. Therefore, using conservation of mass between the engine face and inlet throat, the inlet throat area can be calculated.

$$A_{TH} = A_{EF}\left[\frac{WFF(M_{EF})}{WFF(M_{TH})}\right]\frac{P_{t_{EF}}}{P_{t_{TH}}}\left[1 + \frac{A_{BP}}{A_c} + \frac{A_{VENT}}{A_c}\right]$$

The above relation is used with the appropriate design point input values to calculate the design point inlet throat area.

Inlet Capture Area

The inlet capture area can be computed by using the conservation of mass relation between the inlet throat and the free-stream conditions. The inlet capture area is equal to the free-stream flow area (i.e., $A_o/A_c = 1.0$) at the inlet design point. Therefore,

$$A_c = A_o = A_{TH_D}\left[\frac{WFF(M_{TH})}{WFF(M_{DES})}\right]\left(\frac{P_{t_{TH}}}{P_{t_\infty}}\right)_{DES}\left[1 + \left(\frac{A_{BL}}{A_c}\right)_{DES}\right]$$

ADDITIVE DRAG (MODULE CDADDI)

The additive and subsonic spillage drag computational approach follows Sibulkin (ref. 3). The inputs and outputs of the module are shown in figure 3. If the design Mach number (M_{DES}) is less than or equal to one, the bleed and bypass area ratios, as well as the additive and subsonic spill drags, are set equal to zero. If the design Mach number (M_{DES}) is greater than one, the following are assumed:

1. Axisymmetric cone geometry
2. External compression inlet
3. 20° cone half angle (THETA = 20°) can be varied internally
4. $\cos \lambda = 1.0$
5. Throat Mach number is constant at input value.

The ratio of A_o/A_c for the engine airflow is calculated to be

$$\left(\frac{A_o}{A_c}\right)_{ENG} = \frac{\rho_\infty A_o V_\infty}{\rho_\infty A_c V_\infty} = \frac{W_a}{\rho_\infty A_c V_\infty}$$

where

$$\rho_\infty V_\infty = \frac{WFF(M_\infty)P_{t_\infty}}{(T_{t_\infty})^{1/2}}$$

The bleed and bypass area ratios are then computed from a predetermined schedule which can be changed if desired. The schedules are currently

$$\frac{A_{BL}}{A_c} = 0.10\ \text{SFBEP}\left(\frac{M_{DES}}{3.0}\right)^3\left(\frac{M_\infty - 1.0}{M_{DES} - 1.0}\right)$$

$$\frac{A_{BP}}{A_c} = \text{SFBPP}\left[1.0 - \left(\frac{A_o}{A_c}\right)_{ENG}\right]0.5$$

where

SFBEP = an input scale factor for the bleed flow schedule

SFBPP = an input scale factor for the bypass flow schedule

(Note: If the bleed and/or bypass airflow schedules are changed here, they must also be changed in subroutine SIZIN.)

The ratio of A_o/A_c for the inlet is computed from the engine airflow characteristics and the bleed, bypass, and vent airflow characteristics:

$$\frac{A_o}{A_c} = \left(\frac{A_o}{A_c}\right)_{ENG} (1.0 + \text{WEXWEF})$$

where

$$\text{WEXWEF} = \frac{\rho_\infty A_c V_\infty}{W_a}\left(\frac{A_{BL}}{A_c} + \frac{A_{BP}}{A_c} + \frac{A_{VENT}}{A_c}\right)$$

and

$\frac{A_{VENT}}{A_c}$ is input (0.03 is typical)

The additive drag is computed using Sibulkin's formulation (ref. 3):

$$C_{D_{AD}} = \frac{2}{\gamma M_\infty^2}\left[\frac{A_{TH}}{A_c}\frac{P_{t_\infty}}{P_\infty}\frac{P_{t_{TH}}}{P_{t_\infty}}\frac{P_{TH}}{P_{t_{TH}}}(\gamma M_{TH}^2 + 1)\cos\lambda + \frac{(A_c - A_{TH})}{A_c}\frac{P_{cone}}{P_\infty} - 1.0 - \frac{A_o}{A_c}\gamma M_\infty^2\right] + C_S$$

where

$$\frac{P_{t_\infty}}{P_\infty} = \left(\frac{1 + M_\infty^2}{5}\right)^{3.5}$$

$$\frac{P_{t_{TH}}}{P_{t_\infty}} = PR_{SUP}$$

$$\frac{P_{TH}}{P_{t_{TH}}} = \frac{1}{\left(\frac{1 + M_{TH}{}^2}{5}\right)^{3.5}}$$

$$A_{TH} = \frac{W_a(1.0 + WEXWEF)(T_{t_\infty})^{0.5}}{WFF(M_{TH})PR_{SUP}P_{t_\infty}}$$

For $M_\infty \leq 1.0$, the cone surface Mach number and cone surface pressure ratio are estimated, as follows:

$$M_{cone} \cong M_\infty$$

$$\frac{P_{TH}}{P_\infty} = \left[\frac{1}{(P_\infty/P_{t_\infty})}\right]\left(\frac{P_{t_{TH}}}{P_{t_\infty}}\right)\left(\frac{P_{TH}}{P_{t_{TH}}}\right)$$

where P_{TH} is the cone static pressure at the throat. The cone average pressure is

$$P_{cone} = \frac{\left(\frac{P_{TH}}{P_\infty}\right)P_\infty + P_\infty}{2}$$

and the cone surface pressure ratio is

$$PSPIN = \frac{P_{cone}}{P_\infty}$$

For $M_\infty > 1.0$, the cone surface pressure coefficient can be estimated using an approximation presented by Lighthill (ref. 4):

$$
\begin{aligned}
C_{P_S} &= \text{cone average pressure coefficient} \\
&= -\theta^2 + 2\theta^2 \, ln\left[\frac{2}{(M_\infty^2 - 1)^{1/2}\theta}\right] \\
&\quad + 3(M_\infty^2 - 1)\theta^4 \left\{ ln\left[\frac{2}{(M_\infty^2 - 1)^{1/2}\theta}\right]\right\}^2 \\
&\quad - (5M_\infty^2 - 1)\theta^4 \left\{ ln\left[\frac{2}{(M_\infty^2 - 1)^{1/2}\theta}\right]\right\} \\
&\quad + \left[\frac{13}{4} M_\infty^2 + \frac{1}{2} + \frac{(\gamma + 1)M_\infty^4}{(M_\infty^2 - 1)}\right]\theta^4
\end{aligned}
$$

where θ is the cone half angle in radians.

The cone surface pressure ratio can be obtained from the definition of the pressure coefficient

$$
\text{PSPIN} \equiv P_{cone}/P_\infty = C_{P_S} \times (Q/P_\infty) + 1.0
$$

where

$$
Q/P_\infty = 0.7M_\infty^2
$$

The cone surface Mach number can be approximated by using a formulation of Lighthill (ref. 4):

$$
M_{cone} = \left\{ \frac{M_\infty^2 \left[\frac{1}{4} M_\infty^2 C_{P_S}(\gamma + 1) + 1\right] - M_\infty^2 C_{P_S}\left(\frac{\gamma}{4} M_\infty^2 C_P + 1\right)}{\left[\frac{1}{4} M_\infty^2 C_{P_S}(\gamma - 1) + 1\right]\left(\frac{\gamma}{2} M_\infty^2 C_{P_S} + 1\right)} \right\}^{1/2}
$$

or, for $\gamma = 1.4$,

$$
M_{cone} = M_\infty \left[\frac{(0.6M_\infty^2 C_{P_S} + 1.0) - C_{P_S}(0.35M_\infty^2 C_{P_S} + 1)}{(0.7M_\infty^2 C_{P_S} + 1)(0.1M_\infty^2 C_{P_S} + 1)} \right]^{1/2}
$$

To complete the additive drag calculation, it is necessary to evaluate the subsonic spillage drag C_S. C_S is the drag of the inlet spillage that occurs behind a normal shock. This drag is equal to zero if the free-stream Mach number is subsonic.

Using Sibulkin's formulation (ref. 3), we have

$$C_S = \frac{2}{\gamma M_\infty^2}\left(\frac{A_s - A_y}{A_c}\right)\frac{(\bar{P}/P_{cone} - 1)P_{cone}}{P_\infty}$$

$$A_s = A_c - A_{TH}\cos\lambda \qquad \text{(see fig. 1)}$$

$$A_y = A_c\left\{\left[\left(\frac{A_s}{A_c}\right)^{1/2} - \frac{L\tan\theta}{y_c}\right]\right\}^2$$

$$y_c = \left(\frac{A_c}{\pi}\right)^{1/2}$$

$$\theta = \text{cone half angle} \qquad \text{(see fig. 1)}$$

$$\frac{L}{y_c} = K\left(1.0 - \frac{A_o}{A}\frac{1}{\beta}\right)$$

$$K = f(M_\infty) = 0.2505M_\infty^2 - 1.492625M_\infty + 2.8921$$

(see ref. 3, p. 7)

where β is the ratio of mass flow with supersonic flow at the inlet to the maximum theoretical capture area mass flow.

Note that β is a function of X_{cone}/y_c, M_∞, θ and, according to Sibulkin (ref. 3), β can be considered equivalent "in most cases" to the supercritical mass flow ratio. The supercritical mass flow ratio is presented by Barry (ref. 12) where β is equal to Barry's A_∞/A_o. For the present purposes, it is assumed that

$$\beta = 1.0 \;; \qquad \text{for } X_{cone}/y_c < 1.2$$

$$= 1.0 - (X_{cone}/y_c - 1.2)/(2.75 - 1.2) \;; \qquad \text{for } X_{cone}/y_c \geq 1.2$$

$$\bar{P}/P_{cone} = PNSPC = (7M_{cone}^2 - 1)/6$$

For $M_\infty < 0.4$ or $A_o/A_c > 1.0$,

$$CD_{AD} = 0.0$$

$$C_S \equiv CD_{SP} = 0.0$$

Figure 11 shows a comparison of additive drag coefficient as computed by the methods of reference 3 and by the PRINC program. Sibulkin (ref. 3) assumes the spike position to be a function of M_∞. The PRINC method assumes a spike position that is a function of M_∞ and throttle setting such that the inlet throat Mach number M_{TH} is a constant at the input value.

BYPASS AND BLEED DRAGS

(MODULE CDBYPA)

This module computes the drag coefficients associated with the bypass (CDBP) and bleed (CDBL) systems. The derivation of these drag effects is the same; however, it is usually assumed that the pressure recovery for the bleed system is lower than for the bypass system.

Two assumptions may be made for the bleed and bypass exit nozzles; namely, that they are either (1) sonic nozzles, with $M_E = 1$, or (2) fully expanded nozzles, with

$$P_E = P_\infty$$

$$M_E = \left\{5\left[(P_{T_E}/P_\infty)^{0.286} - 1\right]\right\}^{1/2}$$

The assumption currently used in the bleed and bypass subroutine is that the exit nozzles are sonic; however, if it is desired to use the fully expanded assumption, the changes necessary are contained in subroutine CDBYPA as comment cards. Also, the bleed and bypass drags consider momentum losses only, and do not include any drag that may be associated with the exits themselves. The derivation of the governing equation for the bypass (or bleed) drag is discussed next.

The thrust for the bypass (or bleed) is (see fig. 4)

$$T = (\dot{m}_E V_E + P_E A_E - P_\infty A_E)\cos\theta_E - \dot{m}_{BP} V_\infty$$

where $(\ )_E$ = exit conditions for the bypass and $\dot{m}_{BP} = \dot{m}_E$ from continuity considerations.

The thrust coefficient (based on A_c) is

$$C_D = -C_T = \frac{m_{BP} V_\infty - (\dot{m}_E V_E + P_E A_E - P_\infty A_E)\cos\theta_E}{QA_c}$$

from reference 13,

$$\frac{F}{P} \equiv \frac{\dot{m}V + P(A)}{P} = A(1 + \gamma M^2)$$

$$\frac{f}{p} = \frac{F}{PA} = (1 + \gamma M^2)$$

where F is stream thrust, A is area, and P is static pressure. Using the definition of dynamic pressure,

$$Q = \frac{1}{2}\rho_\infty V_\infty^2 = \frac{1}{2}\gamma M_\infty^2 P_\infty$$

and using the f/p definition, the thrust coefficient can be rewritten

$$C_T = \frac{\cos\theta_E}{(1/2)\gamma M_\infty^2}\left(\frac{f}{p}\right)_E \frac{P_E}{P_{t_E}} \frac{P_{t_E}}{P_{t_\infty}} \frac{P_{t_\infty}}{P_\infty} - \left[\frac{2A_E \cos\theta_E}{\gamma M_\infty^2 A_c} + \frac{A_{BP}}{(1/2)A_c}\right]$$

However,

$$\dot{m}_{BP} \equiv \rho_\infty A_{BP} V_\infty = \rho_E A_E V_E$$

$$\frac{1}{P_\infty} = \frac{P_E}{P_{t_E}} \frac{P_{t_E}}{P_{t_\infty}} \frac{P_{t_\infty}}{P_\infty} \frac{1}{P_E}$$

$$\left(\frac{f}{P}\right)_E \equiv \frac{\dot{m}_E V_E + P_E A_E}{P_E A_E} = (1 + \gamma M_E^2)$$

and, from conservation of energy,

$$T_{t_\infty} = T_{t_E}$$

Using the weight flow function, which, for $\gamma = 1.4$, is

$$WFF(M) = 0.92M\left(\frac{1}{1 + 0.2M^2}\right)^3 = \frac{W_a\sqrt{T_t}}{P_t A}$$

Therefore, the ratio of the exit flow area to the free-stream flow area for the bypass (or bleed) is

$$\frac{A_E}{A_{BP}} = \frac{0.92M_\infty\left(\frac{1}{1 + 0.2M_\infty^2}\right)^3 P_{t_\infty}}{0.92M_E\left(\frac{1}{1 + 0.2M_E^2}\right)^3 P_{t_E}} = \frac{M_\infty}{M_E}\left(\frac{1 + 0.2M_E^2}{1 + 0.2M_\infty^2}\right)^3 \frac{P_{t_\infty}}{P_{t_E}}$$

and thus the thrust coefficient for the bypass (or bleed) is

$$C_T = \frac{\cos\theta}{(1/2)\gamma M_\infty^2}\frac{A_{BP}}{A_c}\left\{\frac{P_{t_\infty}}{P_{t_E}}\frac{M_\infty}{M_E}\left(\frac{1 + 0.2M_E^2}{1 + 0.2M_\infty^2}\right)^3\left[\left(\frac{f}{P}\right)_E\frac{P_E}{P_{t_E}}\frac{P_{t_E}}{P_{t_\infty}}\frac{P_{t_\infty}}{P_\infty} - 1\right]\right\} - 2\frac{A_{BP}}{A_c}$$

or, rearranging terms and using the definition of C_D, gives

$$\frac{C_D}{(A_{BP}/A_c)} = 2\left[1 - \cos\theta_E\frac{M_E}{M_\infty}\left(\frac{1 + 0.2M_\infty^2}{1 + 0.2M_E^2}\right)^{0.5}\right]$$

$$+\left\{\frac{\cos\theta_E}{(\gamma/2)M_\infty^2}\frac{M_\infty}{M_E}\left(\frac{1 + 0.2M_E^2}{1 + 0.2M_\infty^2}\right)^3\left[\frac{1}{(P_{t_E}/P_{t_\infty})} - \left(\frac{1 + 0.2M_\infty^2}{1 + 0.2M_E^2}\right)^{3.5}\right]\right\}$$

Note: The derivation of the bleed drag coefficient is identical to the above derivation with the exception of the appropriate subscripts.

It is currently assumed that

$$\frac{P_{t_{E_{Bleed}}}}{P_{t_\infty}} = 0.3\,\frac{P_{t_{EF}}}{P_{t_\infty}}$$

$$\frac{P_{t_{E_{Bypass}}}}{P_{t_\infty}} = 0.7\,\frac{P_{t_{EF}}}{P_{t_\infty}}$$

It is also assumed that both the bleed and bypass systems have sonic exit nozzles.

Figures 12 and 13 show example calculations of bypass and bleed drag coefficients for sonic exit Mach numbers. Engine face total pressure recovery and bypass and bleed mass flow schedules for a study supersonic transport configuration from reference 14 are presented in figure 12. These values are used as inputs to the PRINC module and the calculated drag coefficients that are based on inlet capture area are shown in figure 13. The bypass results (fig. 13a) of reference 14, and the PRINC module calculations (dashed curve) are based on an exit angle of 10° and on a bypass pressure recovery that is assumed equal to the engine face recovery. The PRINC module results agree well with those of reference 14. A calculated curve from PRINC module that indicates the effects of bypass recovery and exit angle is also shown in figure 13a. PRINC module calculated bleed drag coefficients, shown in

figure 13b, are compared to reference 14 values for a recovery that is three-tenths the engine face recovery. Again, the agreement is good. Also, the effect of changing bleed exit angle on the PRINC module results is indicated in the figure.

AUXILIARY SYSTEMS DRAG

(MODULE CDAUXI)

This module computes the drag coefficient (based on A_c) associated with the auxiliary system (CD_{AUX}) such as losses for cooling air for various equipment and compartments. A description of this drag increment is given in reference 6. For these calculations, the total momentum is assumed lost.

Therefore,

$$C_{D_{AUX}} = \frac{\dot{m}_{AUX} V_\infty}{QA_c} = \frac{\rho_\infty A_{AUX} V_\infty^2}{(1/2)\rho_\infty V_\infty^2 A_c}$$

$$= 2\,\frac{A_{AUX}}{A_c}$$

where A_{AUX}/A_c is a user input and is generally a small value on the order of 0.005 to 0.01.

BOUNDARY-LAYER DIVERTER DRAG

(MODULE CDDIVI)

This module computes the drag coefficient (based on A_c) of the nacelle/airframe boundary-layer diverter system $C_{D_{DIV}}$. A diverter half angle θ_D of 20° is assumed and the ratio of diverter height to boundary-layer height is approximately 0.5. The procedure used is to curve fit the empirical diverter pressure coefficients from two references:

Reference 5, pg. 3-24, gives data at M = 0.9, 1.57 and 1.97.

Reference 6, pg. III.B.4.2, gives data at M = 2.0 and 3.0.

The curve fit yields the following relations:

$$C_{D_{DIV}} = \frac{1.2}{M_\infty^2}\,\frac{\theta_D}{20}\,\frac{A_{WEDGE}}{A_c}\ ; \qquad \text{for } M_\infty \geq 1.55$$

$$= 0.499\,\frac{\theta_D}{20}\,\frac{A_{WEDGE}}{A_c}\ ; \qquad \text{for } 0.95 \leq M_\infty \leq 1.55$$

$$C_{D_{DIV}} = \frac{(M_\infty - 0.8)}{(0.95 - 0.80)} \frac{\theta_D}{20} \frac{A_{WEDGE}}{A_c} \times 0.499 \ ; \qquad \text{for } 0.80 \leq M_\infty \leq 0.95$$

$$= 0.0 \ ; \qquad \text{for } M_\infty \leq 0.8$$

where A_{WEDGE}/A_c is a user input.

Figure 14 shows a comparison of the diverter pressure coefficients computed by the PRINC module with data from the two references for various Mach numbers.

BOATTAIL DRAG (MODULE CDBTA)

The drag on the airframe back to the fuselage end point (the "customer connect" point, see fig. 1b) is calculated as part of the airplane drag. The drag on the portion of the engine nozzle aft of this point is defined as the boattail drag. The boattail drag is a function of the free-stream Mach number, the boattail angle, and the length of the boattail. The performance penalty for this drag is charged to the engine performance in accordance with the ACSYNT bookkeeping system. The boattail drag estimation method used here is described in reference 2. The boattail drag coefficient is based on the nozzle area per engine at the "customer connect" point in reference 2; however, the basis is changed to the inlet capture area per engine in the program. The ratio of nozzle area per engine at the customer connect to inlet capture area per engine required for the change is

$$\frac{A_{CC}}{A_c} = \frac{\pi (D_{CC})^2}{4A_c}$$

The curve fit of drag coefficients based on A_{CC} (from ref. 2, fig. 41) yields

$$C_{D_\beta} = 0.0102 \left(\frac{\beta}{16}\right) \frac{1}{(1 - M_\infty^{1.5})} \ ; \qquad \text{for } M_\infty \leq 0.95$$

$$C_{D_\beta} = \frac{1.4 \tan \beta}{M_\infty^{1.53}} \left[1 - \left(\frac{D_g}{D_{CC}}\right)^2\right] ; \qquad \text{for } M_\infty \geq 1.0$$

For Mach numbers between 0.95 and 1.0, interpolate linearly between the above relations. These equations are for a nozzle pressure ratio of 2.5.

Values for the above equations are

$$D_{ENG} = \sqrt{\frac{4A_{ENG}}{\pi}}$$

where A_{ENG} is an input from the engine calculation.

$$D_{CC} = 1.10\ D_{ENG}$$

$$M_{EXIT} = \left[\frac{\left(\dfrac{P_\infty}{P_{t_{NOZ}}}\right)^{-\left(\frac{\gamma-1}{\gamma}\right)} - 1}{\dfrac{\gamma-1}{2}}\right]^{1/2} \quad ; \qquad \text{(ref. 11)}$$

where $P_\infty/P_{t_{NOZ}} = 1/NPR$ which is input.

$$A_{NOZ_{TH}} = \frac{1}{WFF(1)} \frac{\sqrt{T_{t_{NOZ}}}}{P_{t_{NOZ}}} W_a$$

where WFF(1) is the weight flow function at $M = 1.0$; $T_{t_{NOZ}}$, $P_{t_{NOZ}}$, and W_a are input.

$$\frac{A_{NOZ_{TH}}}{A_{EXIT}} = \left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{2(\gamma-1)}} M_{EXIT} \left(1 + \frac{\gamma-1}{2} M_{EXIT}^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}} \quad ; \qquad \text{(see ref. 11)}$$

$$A_{EXIT} = \frac{A_{NOZ_{TH}}}{A_{NOZ_{TH}}/A_{EXIT}}$$

$$D_g = \sqrt{\frac{4A_{EXIT}}{\pi}}$$

Assume $L_{NOZ} = D_{ENG}$, then

$$\beta' = \tan^{-1}\left(\frac{D_{CC} - D_g}{2L_{NOZ}}\right) \text{ in radians}$$

$$\beta = 57.3 \times \beta' \text{ in degrees}$$

To correct for nozzle pressure ratio NPR, which is an input value from the engine calculation, use reference 2, figure 42:

$$\Delta C_{D_\beta} = 0 \; ; \quad \text{if NPR} \le 3$$

$$\Delta C_{D_\beta} = 0.005(\text{NPR} - 3) \; ; \quad \text{if NPR is between 3 and 4}$$

$$\Delta C_{D_\beta} = 0.01(\text{NPR} - 4) + 0.005 \; ; \quad \text{if NPR is between 4 and 8}$$

and

$$\Delta C_{D_\beta} = 0.045 \; ; \quad \text{if NPR} \ge 8$$

The corrected C_{D_β} is then

$$C_{D_\beta} = C_{D_{\beta_{2.5}}} - \Delta C_{D_\beta}$$

To base coefficient on capture area,

$$C_{D_{BT}} = C_{D_\beta}\left(\frac{A_{CC}}{A_c}\right)$$

where $A_{CC} = \pi D_{CC}^2/4$, as previously described. Finally, if

$$C_{D_{BT}} \le 0 \; , \; \text{set} \; C_{D_{BT}} = 0 \; .$$

Figure 15a is a plot of the PRINC module computed $C_{D_{BT}}$ (based on a customer-connect area of 3 ft^2) for a nozzle pressure ratio of 2.5 and for two different boattail angles. Data from reference 2 is also shown (symbols) for the same conditions, indicating the ACSYNT calculations are low for Mach numbers below about 0.8.

A comparison of Boeing lightweight fighter data (ref. 2) and PRINC module calculations for the same nozzle (based on a reference area of 20.2 ft^2) is shown in figure 15b. The nozzle pressure ratio for the data is not known, so several values are shown for the calculations. The PRINC module overpredicts at supersonic speeds and underpredicts at subsonic Mach numbers for this nozzle configuration.

NOZZLE INTERFERENCE DRAG (MODULE ENGCDI)

This module calculates the interference drag on the base between multiple nozzle afterbodies. The procedure used is an interpolation between the curves (C_{D_I}') of reference 2, figure 46, which have been tabularized and put into the program. C_{D_I}' is the interference drag coefficient between two engines for a nozzle pressure ratio of 2.5. The independent variables are Mach number (M_∞)

and nozzle spacing ratio S/D_g, where S is the distance between adjacent nozzle centerlines and D_g is the jet diameter. The value S/D_g is a user input to the program. The final interference drag coefficient $C_{D_{INF}}$ is based on capture area per engine. For a given S/D_g and M_∞, C_{D_I}' is obtained from the table look up for a nozzle pressure ratio of 2.5. To determine the final $C_{D_{INF}}$ for any given nozzle pressure ratio and capture area, the following correction is applied:

$$C_{D_{INF}} = \left(\frac{2.5}{NPR}\right)\left(\frac{N_{ENG} - 1}{N_{ENG}}\right)\left(\frac{C_{D_I}' \times 2 \times T_g}{QA_c}\right)$$

where 2.5/NPR is a correction for nozzle pressure ratio and NPR is input to the program from the engine calculation; $(N_{ENG} - 1)/N_{ENG}$ is a correction for number of engines since desired output is per engine and N_{ENG} is input to the program; and T_g is gross thrust per engine for the given M_∞ and power setting and is input to the program from the engine calculation.

Figure 16 is a plot ACSYNT determined $C_{D_{INF}}$ for several values of S/D_g compared with the data of reference 2. The graph simply shows the accuracy of the table look up procedures while giving an indication of the magnitude and variation of the results with Mach number.

CONTROL ROUTINE (XINLET)

This portion of the program controls the sequence of calling the various modules. In addition, it converts all the drag coefficients to the wing reference area and to the proper number of engines, since the values from the various modules are based on capture area per engine.

APPENDIX B

MODULE LISTING

This appendix contains the FORTRAN listing for the Propulsion Installation Calculation (PRINC) module for the ACSYNT program.

```
      SUBROUTINE CDBYPA(XMO,ABYPAC,ABLEAC,CDBE,CDBP,PRTOT,PINF,PTO)
C
C     COMPUTES THE BYPASS AND BLEED EFFECTS
C     THE ADDITIVE DRAG CALCULATION IS FOR THE TOTAL AIRFLOW
C     ENTERING THE INLET.  USING THIS BOOKKEEPING THE EFFECT OF
C     THE BYPASS AND BLEED MUST BE ADDED IN.
C     XMO=FREE STREAM MACH NO
C     ABYPAC=ABYPASS/AC  AT FREESTREAM
C     ABLEAC=ABLEED/AC  AT FREESTREAM
C     CDBE=INCREMENTAL DRAG COEF FOR BLEED BASED ON AC
C     CDBP=INCREMENTAL DRAG COEF FOR BYPASS BASED ON AC
C     PRTOT=INLET TOTAL PRESSURE RECOVERY TO ENGINE FACE
C     PINF=FREESTREAM STATIC PRESSURE (PSF)
C     PTO=FREESTREAM TOTAL PRESSURE (PSF)
C     PTBPPT=BYPASS TOTAL PRESSURE RECOVERY (.7*ENGINE FACE PRES REC)
C     PTBEPT=BLEED TOTAL PRESSURE RECOVERY (.3*ENGINE FACE PRES REC)
C
      QPT(XM)=.7*XM*XM*(1.+.2*XM*XM)**(-3.5)
      FPT(XM)=(1.+1.4*XM*XM)*(1.+.2*XM*XM)**(-3.5)
      PPT(XM)=(1.+.2*XM*XM)**(-3.5)
C
C     ASSUME EXIT ANGLE FOR BLEED AND BYPASS = 15 DEG
C
      XMO2=XMO*XMO
      COSDE=.966
      PTBPPT=.7*PRTOT
      PTBYP=PTBPPT*PTO
      PTBPIN=PTBYP/PINF
      IF (PTBPIN.GT.1.) GO TO 10
      CDBP=0.
      GO TO 20
C
C     ASSUME THE BYPASS EXIT IS FULLY EXPANDED.  FOR A SONIC
C     BYPASS NOZZLE SET XMEBY (NOZZLE EXIT MACH NO) = 1.
C      XMEBY=SQRT(5.*((PTBYP/PINF)**.286-1.))
C     ASSUME A SONIC EXIT
C
   10 XMEBY=1.
      XMEMOR=(1.+.2*XMEBY*XMEBY)/(1.+.2*XMO*XMO)
      CDBP2=2.*(1.-((XMEBY/XMO)*(XMEMOR**(-.5))*COSDE))
      CDBP2=CDBP2+(COSDE/(.7*XMO2)*XMO/XMEBY*(XMEMOR**3)*(1./PTBPPT
     1 -(XMEMOR**(-3.5))))
      CDBP=CDBP2*ABYPAC
      IF (CDBP.LE.0.) CDBP=0.
C
C
C     FOR BLEED
C
   20 PTBEPT=.3*PRTOT
      PTBLE=PTBEPT*PTO
      PTBLPI=PTBLE/PINF
      IF (PTBLPI.GT.1.) GO TO 30
      CDBE=0.
      RETURN
C
C     ASSUME THE BYPASS EXIT IS FULLY EXPANDED.  FOR A SONIC
```

```
C       BLEED NOZZLE SET XMEBE (BLEED NOZZLE EXIT MACH) = 1.
C        XMEBE=SQRT(5.*((PTBLE/PINF)**.286-1.))
C       ASSUME A SONIC EXIT NOZZLE FOR THE BLEED
C
   30 XMEBE=1.
      XMEMOE=(1.+.2*XMEBE*XMEBE)/(1.+.2*XMO2)
      CDBE2=2.*(1.-((XMEBE/XMO)*(XMEMOE**(-.5))*COSDE))
      CDBE2=CDBE2+(COSDE/(.7*XMO2)*XMO/XMEBE*(XMEMOE**3)*(1./PTBEPT
     1 -(XMEMOE**(-3.5))))
      CDBE=CDBE2*ABLEAC
      IF (CDBE.LE.0.) CDBE=0.
      IF (CDBP.LE.0.) CDBP=0.
      RETURN
      END
```

```
      SUBROUTINE PRSUBS(XMT,PRSUB)                                      PRSU0010
C                                                                       PRSU0020
C     COMPUTES THE SUBSONIC DIFFUSER PRESSURE RECOVERY                  PRSU0030
C                                                                       PRSU0040
      XMT2=XMT*XMT                                                      PRSU0050
      EPS=.37148*XMT2-.231428*XMT+.05                                   PRSU0060
      PRSUB=1.-EPS*(1.-1./(1.+.2*XMT2)**3.5)                            PRSU0070
      RETURN                                                            PRSU0080
      END                                                               PRSU0090
```

```
      SUBROUTINE CDBTA(XMO,XNPR,PTNOZ,TTNOZ,AENG,CDBT,WA,AC,AEXIT,BETA)
C
C     XMO=FREE STREAM MACH NUMBER
C     XNPR=NOZZLE PRESSURE RATIO
C     PTNOZ=NOZZLE EXIT TOTAL PRESSURE
C     TTNOZ=NOZZLE EXIT TOTAL TEMPERATURE
C     AENG=ENGINE FACE TOTAL AREA, SQ FT
C     CDBT=BOATTAIL DRAG PER ENGINE, REFERENCED TO AC
C     WA=ENGINE AIRFLOW, LBS/SEC
C     AC=INLET CAPTURE AREA, SQ FT
C     CORNPR=DRAG CORRECTION FACTOR FOR NPR
C     AEXIT=NOZZLE EXIT AREA PER ENGINE, SQ FT
C
      WFF(XM)=.92*XM/(1.+.2*XM*XM)**3
      XMEX=SQRT((XNPR**.286-1.)/.2)
      ANOZT=SQRT(TTNOZ)/PTNOZ*WA/WFF(1.)
      ANTAE=1.728*XMEX/(1.+.2*XMEX*XMEX)**3
      AEXIT=0.
      IF (ANTAE.GT.0.) AEXIT=ANOZT/ANTAE
C
C     ASSUME
C        A CUSTOMER CONNECT = 1.21*AENG
C        LNOZ = DIA OF ENG
C
C     LIMIT THE MAX EXHAUST DIAMETER TO CUSTOMER CONNECT DIAMETER
C     ACC = AREA AT CUSTOMER CONNECT POINT, SQ FT
C
      ATESI=1.21*AENG
      IF (AEXIT.GT.ATESI) AEXIT=ATESI
      DENG=2.*SQRT(AENG/3.14159)
      DCC=1.1*DENG
      DEXIT=2.*SQRT(AEXIT/3.14159)
      DEXDCC=DEXIT/DCC
      AEXIT=ATESI
      XLNOZ=DENG
      BETAI=ARSIN((DCC-DEXIT)/(2.*XLNOZ))
      IF (BETAI.LT.0.) BETAI=0.
      BETA=57.2957795*BETAI
      TBI=1.4*TAN(BETAI)
      CONTBI=TBI*(1.-DEXDCC*DEXDCC)
      IF (XMO.LE..95) CDBT=.0102/(1.-XMO**1.5)*BETA/16.
      IF (XMO.GE..95) CDBT1=CONTBI/XMO**1.53
      IF (XMO.LT..95.OR.XMO.GT.1.) GO TO 10
      CDBT2=CONTBI
      CDBT=CONTBI/.95**1.53
      CDBT=CDBT+20.*(CDBT2-CDBT)*(XMO-.95)
   10 IF (XMO.GE.1.) CDBT=CDBT1
C
C     SET ALL CDBT > (CDBT AT M=1.2) EQUAL TO (CDBT AT M=1.2)
C     THEN CORRECT BOATTAIL DRAG DUE TO XNPR VARIATIONS
C
      CDBT12=CONTBI/1.2**1.53
      IF (CDBT12.LT.CDBT) CDBT=CDBT12
      CORNPR=0.
      IF (XNPR.GT.3..AND.XNPR.LT.4.) CORNPR=.005*(XNPR-3.)
      IF (XNPR.GE.4..AND.XNPR.LE.8.) CORNPR=.01*(XNPR-4.)+.005
```

```
      IF (XNPR.GT.6.) CORNPR=.045
      CDBT=CDBT-CORNPR
      ACC=.785398*DCC*DCC
      CDBT=CDBT*ACC/AC
      IF (CDBT.LT.0.) CDBT=0.
      RETURN
      END
```

```
      SUBROUTINE CDDIVI (XMD,AWAENG,CDDIV)
C
C     FIT DATA AT M=2. IN G/D HBK PG 3.8.4.2
C     ASSUME DIVERTER HEIGHT = .5
C     FIT DATA AT M=.9 IN INT AERO MANUAL (N/A) PG 3-24
C     THDIV = DIVERTER INCLUDED ANGLE
C     AWAENG = AREA OF DIVERTER WEDGE DIVIDED BY AC
C
      THDIV=20.
      CDDIV=0.
      IF (XMD.GT..8.AND.XMD.LT..95) CDDIV=(XMD-.8)/.15*.499*THDIV/20.
      IF (XMD.GE..95.AND.XMD.LT.1.55) CDDIV=.02495*THDIV
      IF (XMD.GE.1.55) CDDIV=.06*THDIV/(XMD*XMD)
      CDDIV=CDDIV*AWAENG
      RETURN
      END
```

```
      SUBROUTINE SIZIN(AEF,XMEF,XMT,PRDES,AC,XMDES,PRSUB,SFBEP)
C
C     SUBROUTINE TO SIZE INLETS
C
C     DESIGN BLEED AND BYPASS SCHEDULES FOR INLET
C     NOTE THAT THESE SCHEDULES MUST BE COMPATIBLE WITH THE
C     BLEED AND BYPASS SCHEDULES IN SUBROUTINE CDADDI
C     AEF = ENGINE FACE FLOW AREA, FT*FT
C     XMEF = ENGINE FACE FLOW MACH NO
C     XMT = THROAT MACH NO
C     PR = SUPERSONIC DIFF. P.R.
C     PRSUB = SUBSONIC DIFF. P.R.
C     ATD = DESIGN THROAT FLOW AREA, FT*FT
C     AT = THROAT FLOW AREA, FT*FT
C     AC = INLET CAPTURE AREA, FT*FT
C     XMDES = INLET DESIGN MACH
C     PRTOT = TOTAL PRESS REC. TO EF
C     SFBEP = SCALE FACTOR FOR INLET BLEED DRAG
C     SFBPP = SCALE FACTOR FOR INLET BYPASS DRAG
C
      WFF(XM)=.92*XM/(1.+.2*XM*XM)**3
      ABLACD=.1*(XMDES/3.)**3*SFBEP
      AVEACD=.03
      WFFXMT=WFF(XMT)
      ATD=AEF*(WFF(XMEF)/WFFXMT)*PRSUB*(1.+AVEACD)
      AC=ATD*(WFFXMT/WFF(XMDES))*PRDES*(1.+ABLACD)
      IF (AC.LE.ATD) AC=ATD
      RETURN
      END
```

```
      SUBROUTINE ENGCDI(XMO,EN,SODG,CDI,FIP,Q,AC,XNPF)
      DIMENSION SD55(4),CD55(4),SD75(4),CD75(4),SD85(4),CD85(4),SD95(4),
     1 CD95(4),SD10(4),CD10(4)
      DATA SD55/1.,1.4,1.8,3.5/,CD55/.005,.0115,2*.0025/
      DATA SD75/1.,1.5,1.8,3.5/,CD75/.005,.013,2*.005/
      DATA SD85/1.,1.7,2.1,3.5/,CD85/.006,.018,2*.006/
      DATA SD95/1.,2.2,2.6,3.5/,CD95/.007,.0375,2*.025/
      DATA SD10/1.,2.3,2.7,3.5/,CD10/.008,.062,.035,.028/
      CDI=0.
      IF (EN.LE.1..OR.XMO.LE.0.) RETURN
      IF (XMO.LT.1.2) GO TO 30
      IF (XMO.LT.1.8) GO TO 10
C
C     XMO GREATER THAN OR EQUAL TO 1.8
C
      CDI1=.005
      GO TO 100
C
C     XMO BETWEEN 1.2 AND 1.8
C
   10 IF (XMO.GT.1.5) GO TO 20
      DM=(XMO-1.2)/.3
      CDI1=.019-.0115*DM
      GO TO 100
   20 DM=(XMO-1.5)/.3
      CDI1=.0075-.0025*DM
      GO TO 100
C
C     XMO LESS THAN 1.2, TABLE LOOKUP REQUIRED
C
   30 DMONU=0.
      DMONL=0.
      IF (XMO.GT.1.) GO TO 80
      IF (XMO.GT..95) GO TO 70
      IF (XMO.GT..85) GO TO 60
      IF (XMO.GT..75) GO TO 50
      IF (XMO.GT..55) GO TO 40
C
C     XMO BETWEEN 0. AND .55
C
      CALL TAINT(SD55,CD55,SODG,CDU,4,1,NERR,DMONU)
      CDL=0.
      DMTAB=.55
      DM=XMO
      GO TO 90
C
C     XMO BETWEEN .55 AND .75
C
   40 CALL TAINT(SD75,CD75,SODG,CDU,4,1,NERR,DMONU)
      CALL TAINT(SD55,CD55,SODG,CDL,4,1,NERR,DMONL)
      DMTAB=.2
      DM=XMO-.55
      GO TO 90
C
C     XMO BETWEEN .75 AND .85
C
```

```
   50 CALL TAINT(SD85,CD85,SDDG,CDU,4,1,NERR,DMONU)
      CALL TAINT(SD75,CD75,SDDG,CDL,4,1,NERR,DMONL)
      DMTAB=.1
      DM=XMD-.75
      GO TO 90
C
C     XMD BETWEEN .85 AND .95
C
   60 CALL TAINT(SD95,CD95,SDDG,CDU,4,1,NERR,DMONU)
      CALL TAINT(SD85,CD85,SDDG,CDL,4,1,NERR,DMONL)
      DMTAB=.1
      DM=XMD-.85
      GO TO 90
C
C     XMD BETWEEN .95 AND 1.0
C
   70 CALL TAINT(SD10,CD10,SDDG,CDU,4,1,NERR,DMONU)
      CALL TAINT(SD95,CD95,SDDG,CDL,4,1,NERR,DMONL)
      DMTAB=.05
      DM=XMD-.95
      GO TO 90
C
C     XMD BETWEEN 1.0 AND 1.2
C
   80 CDU=.019
      CALL TAINT(SD10,CD10,SDDG,CDL,4,1,NERR,DMONL)
      DMTAB=.2
      DM=XMD-1.
   90 CDI1=CDL+(CDU-CDL)*DM/DMTAB
C
C     DETERMINE CDI FOR THE ENGINES
C
  100 CDIT=(2.5/XMPR)*CDI1*2.*FIP/(Q*AC)
      CDI=(EN-1.)*CDIT/EN
      RETURN
      END
```

```
      SUBROUTINE CDADDI(AT,AC,XMO,PR,XMT,XMDES,PTO,TTO,CDAD,CDADS,
     1 ABYPAC,ABLEAC,AOAC,WA,SFBEP,SFBPP)
C
C      AT = INLET THROAT FLOW AREA, FT*FT
C      AC = INLET CAPTURE AREA, FT*FT
C      XMO = FREE STREAM MACH
C      PR = SUPERSONIC DIFFUSER TOTAL PRESSURE RECOVERY
C      XMT = INLET THROAT MACH NO
C      XMDES = INLET DESIGN MACH NO
C      PTO = FREE STREAM TOTAL PRESSURE
C      TTO = FREE STREAM TOTAL TEMP
C      CDAD = SUPERSONIC SPILL ADDITIVE DRAG BASED ON AC
C      CDADS = SUBSONIC SPILL ADDITIVE DRAG BASED ON AC
C      AOAC = AO/AC
C      ABLEAC = ABLE/AC
C      ABYPAC = ABYP/AC
C      WA = ENGINE AIRFLOW, LBS/SEC
C      XMCONE = CONE SURFACE MACH NUMBER
C      CPCS = CONE SURFACE PRESSURE RECOVERY
C      PSPIN = PSTATIC ON CONE/PSTATIC FREE STREAM
C      PNSPC = STATIC PRESS. RATIO ACROSS N.S. AT CONE SURF MACH
C
      PPTO1(XM)=(1.+.2*XM*XM)**(-3.5)
      WFF(XM)=.52*XM/(1.+.2*XM*XM)**3+.0001
      CINPIN(XM)=.00001+.7*XM*XM
      PNSPCO(XM1)=(7.*XM1*XM1-1.)/6.
      WFFXMO=WFF(XMO)
      AOACG=WA*SQRT(TTO)/(WFFXMO*PTO*AC)
      AOAC=AOACG
      CS=0.
      XMOS=XMO*XMO
      IF (XMDES.GT.1.) GO TO 10
      CDAD=0.
      CDADS=0.
      ABYPAC=0.
      ABLEAC=0.
      RETURN
   10 GAM=1.4
      THETA=20.
      THETA1=THETA/57.2957795
      COSLAM=1.
C
C      THE NEXT CARDS ARE THE BLEED AND BYPASS SCHEDULES FOR THE INLET
C      THESE SHOULD BE MADE COMPATIBLE WITH THE INLET DESIGN POINT
C      VALUES IN SUBOUTINE SIZIN
C
      ABLEAC=.1*SFBEP*(XMDES/3.)**3*(XMO-1.)/(XMDES-1.)
      ABYPAC=.5*SFBPP*(1.-AOACG)
      IF (XMO.GT.1.) GO TO 20
      CDAD=0.
      ABLEAC=0
      XMCONE=(XMO+XMT)/2.
      PSPIN=1.
   20 IF (AOACG.GT..97) ABYPAC=0.
      AVEAC=.03
      WCAP=AC*WFFXMO*PTO/SQRT(TTO)
```

```
      WEXWEF=(ABLEAC+ABYPAC+AVEAC)*WCAP/WA
      ADAC=ADACG*(1.+WEXWEF)
C
C     INLET GEOMETRY
C
      RHOTVT=WFF(XMT)*PR*PTO
      AT=WA*(1.+WEXWEF)*SQRT(TTO)/RHOTVT
      IF (AT.GE.AC) AT=.99*AC
      AS=AC-AT*COSLAM
      RCONE=SQRT(AS/3.1416)
      XACONE=RCONE/(TAN(THETA1))
      YC=SQRT(AC/3.1416)
      XCOYC=XACONE/YC
C
C     NOTE THAT THE INLET GEOMETRY HAS BEEN SPECIFIED BY ASSUMING THE
C     THROAT MACH NUMBER AND THE ANGLE OF THE CONE OR RAMP
C     AT = INLET THROAT FLOW AREA, FT*FT
C     AS = FRONTAL AREA OF INLET C/B AT INLET THROAT, FT*FT
C     RCONE = RADIUS OF CONE AT INLET THROAT, FT
C     XACONE = DISTANCE FROM CONE TIP TO INLET THROAT, FT
C     YC = RADIUS CORRESPONDING TO INLET CAPTURE AREA, FT
C     XCOYC = XCONE/YC
C
C     FOR RAMP OF CONE STATIC PRESSURE USE AVERAGE OF
C     FREE STREAM AND THROAT STATIC PRESSURE
C
      P1PO=PR*PPTOT(XMT)/PPTOT(XM0)
      CPCS=.5*(P1PO-1.)/QINPIN(XM0)
      PSPIN=(P1PO+1.)/2.
      IF (XM0.LE.1.) GO TO 30
C
C     CONE SURFACE PRESSURE RATIO
C
      XM011=XM01-1.
      SQXM01=SQRT(XM011)
      ALG=ALOG(2./(SQXM01*THETA1))
      THETA4=THETA1**4
      A1=THETA1**2*(2.*ALG-1.)
      A2=3.*XM011*THETA4*ALG*ALG
      A3=-(5.*XM01-1.)*THETA4*ALG
      A4=THETA4*(13.*XM01/4.+.5+2.4*XM01*XM01/XM011)
      CPCS=A1+A2+A3+A4
      IF (CPCS.GT..9) CPCS=.9
      PSPIN=CPCS*QINPIN(XM0)+1.
   30 IF (XM0.GT..4) GO TO 40
      CDAD=0.
      CDADS=0.
      RETURN
   40 CDA=2./(GAM*XM01)
      CDA1=AT/AC*P1PO*(GAM*XMT*XMT+1.)*COSLAM
      CDA2=(AS/AC)*PSPIN-1.-ADAC*GAM*XM01
C
C     THE APPROX FOR THE CONE SURFACE PRESSURE COEFF INTRODUCES AN
C     ERROR IN THE ADDITIVE DRAG CAL WHICH CAUSES THE ADDITIVE DRAG
C     TO BE NON ZERO AT ADAC=1.; THE NEXT CARDS INTRODUCE A
C     CORRECTION TO THE ADDITIVE DRAG TO COMPENSATE FOR THIS ERROR
```

```
C
      AC1=AC
      ATOD=WFFXMO/WFF(XMT)/PR*AC1
      AS1=AC-ATOD*COSLAM
      CDAD21=(AS1/AC)*PSPIN-1.-GAM*XM01
      CDAD11=ATOD/AC*P1P0*(GAM*XMT*XMT+1.)*COSLAM
      CDCOR=-CDA*(CDAD11+CDAD21)
      IF (XMO.LE.1.) GO TO 50
      XK=.2505*XM01-1.492625*XMO+2.8921
C
C     BETA = AO/AC FOR MINIMUM SUPERCRITICAL SPILLAGE
C     BETA = FCN(XCOYC,XMO,THETA)
C     FOR PRESENT USE APPROX VALUE FOR 20 DEG CONE AND XMO=1.4
C
      BETA=1.
      IF (XCOYC.GT.1.2) BETA=1.-(XCOYC-1.2)/1.55
      IF (BETA.LE.0.) BETA=.00001
      TANTH=TAN(THETA1)
      XLYC=XK*(1.-ADAC/BETA)
      AY=AC*(SQRT(AS/AC)-XLYC*TANTH)**2
C
C     CONE SURFACE MACH NUMBER
C
      B1=XM01*CPCS
      B2=.6*B1+1.
      B3=.35*B1+1.
      B4=.7*B1+1.
      B5=.1*B1+1.
      XMCONE=XMO*SQRT((B2-CPCS*B3)/(B4*B5))
      IF (XMCONE.LT.1.) XMCONE=1.
      PNSPC=PNSPCC(XMCONE)
      CS=(2./(GAM*XM01))*((AS-AY)/AC)*(PNSPC-1.)*PSPIN
   50 CDAD=CDA*(CDA1+CDA2)+CDCOR
C
C     WONG AND ANDERSON FORMULATION OF ADDITIVE DRAG
C     REF NASA TN D-7445
C
      V1VO=(.2*XMT/XMO)*SQRT((1.+.2*XM01)/(1.+XMT*XMT))
      CPST1=(P1P0-1.)/QINPIN(XMO)
      CDADZ=2.*ADAC*(V1VO*COSLAM-1.)+CPST1*AT*COSLAM/AC+CPCS*AS/AC
      CDADZ=CDADZ+CDCOR
      CDADS=CS
      IF (ADAC.LT.1.) GO TO 60
      CDAD=0.
      CDADS=0.
      RETURN
   60 IF (CDAD.LT.0.) CDAD=0.
      IF (CDADS.LT.0.) CDADS=0.
      RETURN
      END
```

```
      SUBROUTINE XINLET(PT0,TT0,PTNOZ,TTNOZ,Q,FIP,WAC,PINF,AENG,PRSUB,  XINL0001
     1 PR,PRTOT,AT,KEYZ)                                                 XINL0002
      COMMON/PRCOM/Z1(3),EN,Z2(15),SWING,Z3(2),CDINSP,Z4(66)            XINL0003
      COMMON/PROC/XMEF,Z5(15),XM0,Z6(7),AEF,Z7(9)                        XINL0004
      COMMON/PRJINS/XNPR,DELPR,IPR,XMT,XMDES,AUAENG,AWAENG,SODG,         XINL0005
     1 XMPRI(6),XPRI(6),Y1,CDAFTP,Y2,CDINLP,Y3,CDIP,CDBTP,CDDIVP,CDAUXP,XINL0006
     2 CDBPP,CDBLP,CDADSP,CDADP,ADAC,Y4(3),PCDFAC,SFINSP,SFADP,SFADSP,  XINL0007
     3 SFBLP,SFBPP,SFAUXP,SFDIVP,SFIP,SFBTP,ABLEAC,ABYPAC,BETA          XINL0008
      COMMON/STDRAG/Y5(4),AC,ANOZ,Y6(14)                                 XINL0009
C                                                                        XINL0010
C     ROUTINE TO CALCULATE INSTALLATION LOSSES                           XINL0011
C     XM0     = FREE STREAM MACH NUMBER                                  XINL0012
C     DELPR   = INCREMENTAL PRESSURE RECOVERY REDUCTION INPUT            XINL0013
C               AS A POSITIVE NUMBER                                     XINL0014
C     IPR     = PRESSURE RECOVERY CODE                                   XINL0015
C     AEF     = ENGINE FACE FLOW AREA, FT*FT                             XINL0016
C     XMEF    = ENGINE FACE MACH NUMBER                                  XINL0017
C     XMT     = THROAT MACH NUMBER                                       XINL0018
C     XMDES   = INLET DESIGN MACH NUMBER                                 XINL0019
C     PT0     = FREE STREAM TOTAL PRESSURE                               XINL0020
C     TT0     = FREE STREAM TOTAL TEMPERATURE                            XINL0021
C     AUAENG  = AUXILIARY AREA OVER AC                                   XINL0022
C     AWAENG  = DIVERTER WEDGE AREA OVER AC                              XINL0023
C     XNPR    = NOZZLE PRESSURE RATIO                                    XINL0024
C     PTNOZ   = NOZZLE EXIT TOTAL PRESSURE                               XINL0025
C     TTNOZ   = NOZZLE EXIT TOTAL TEMPERATURE (R)                        XINL0026
C     SODG    = NOZZLE SPACING OVER JET DIAMETER                         XINL0027
C     Q       = FREE STREAM DYNAMIC PRESSURE                             XINL0028
C     FIP     = ENGINE GROSS THRUST PER ENGINE                           XINL0029
C     WAC     = ENGINE CORRECTED WEIGHT FLOW                             XINL0030
C     PINF    = FREE STREAM STATIC PRESSURE                              XINL0031
C     AENG    = ENGINE FACE TOTAL AREA FT*FT                             XINL0032
C     PRSUB   = INLET SUBSONIC DIFFUSER PRESSURE RECOVERY                XINL0033
C     PR      = INLET SUPERSONIC DIFFUSER PRESSURE RECOVERY              XINL0034
C     PRTOT   = INLET TOTAL PRESSURE RECOVERY TO ENGINE (=PR*PRSUB)      XINL0035
C     AT      = INLET THROAT FLOW AREA (FT*FT)                           XINL0036
C     AC      = INLET CAPTURE AREA (FT*FT)                               XINL0037
C     CDADP   = INLET ADDITIVE DRAG PER A/C BASED ON WING AREA           XINL0038
C     CDADSP  = INLET SUBSONIC SPILL DRAG (PER A/C)   SWING REF          XINL0039
C     CDBLP   = INLET BLEED DRAG (PER A/C) SWING REF                     XINL0040
C     CDBPP   = INLET BYPASS DRAG (PER A/C)  SWING REF                   XINL0041
C     CDAUXP  = INLET AUXILARY AIR DRAG (PER A/C)  SWING REF             XINL0042
C     CDDIVP  = INLET DIVERTER DRAG (PER A/C)   SWING REF                XINL0043
C     CDBTP   = NOZZLE BOATTAIL DRAG (PER A/C) SWING REF                 XINL0044
C     CDIP    = BASE DRAG FOR SPACE BETWEEN ENGINES (PER A/C)  SWING REFXINL0045
C     CDINLP  = INLET TOTAL DRAG PER A/C  SWING REF                      XINL0046
C     CDAFTP  = AFT END TOTAL DRAG PER A/C  SWING REF                    XINL0047
C     CDINSP  = TOTAL PROP INSTALLIATION DRAG PER A/C   SWING REF        XINL0048
C     ADAC    = FREESTREAM FLOW AREA/AC                                  XINL0049
C     ABYPAC  = FREESTREAM FLOW AREA FOR BYPASS/AC                       XINL0050
C     ABLEAC  = FREESTREAM FLOW AREA FOR INLET BLEED/ AC                 XINL0051
C     KEYZ    = 1, INLET DRAG-PR COMPUTED; =2, AFT END EFFECTS COMPUTED XINL0052
C     PCDFAC  = SCALE FACTOR FOR INLET CAPTURE AREA                      XINL0053
C     SF<<<<< = SCALE FACTOR FOR VARIOUS DRAGS                           XINL0054
C               <<<< CORRESPONDS TO INSP, ADP, ETC.                      XINL0055
C     ANOZ    = NOZZLE EXIT AREA, FT*FT                                  XINL0056
```

```
C
      CDADP=0.
      CDADSP=0.
      CDBEP=0.
      CDBPP=0.
      CDAUXP=0.
      CDDIVP=0.
      CDBTP=0.
      CDIP=0.
      CDINLP=0.
      CDAFTP=0.
      CDINSP=0.
      Q=.7*XMO*XMO*PINF
      IF (KEYZ.EQ.2) GO TO 10
      PRSUB=1.
C
C     THE INLET IS ASSUMED TO HAVE GEOMETRY THAT CAN BE VARIED IN
C     SUCH A MANNER TO KEEP THE THROAT MACH NUMBER AT THE INPUT
C     VALUE. THE INLET MODEL IS (FOR THE PRESENT) FOR AN EXTERNAL
C     COMPRESSION AXISYMETRIC INLET. IF IT IS DESIRED TO MODIFY THE CODE
C     TO HANDLE A FIXED INLET THE THROAT FLOW AREA SHOULD BE SET
C     EQUAL TO THE DESIGN VALUE AND THE SUBSONIC DIFFUSER PRESSURE
C     RECOVERY SHOULD BE COMPUTED FROM CONTINUITY.
C
      IF (IPR.GT.10) CALL PRSUBS(XMT,PRSUB)
      CALL PRINL(XMO,DELPR,PR,IPR,PRTOT,PRSUB,XMPRI,XPRI,PRDES,XMDES)
      IF (IPR.LT.0) RETURN
      WA=.010795*WAC*PTO*PRTOT/SQRT(TTO)
      CALL SIZIN(AEF,XMEF,XMT,PRDES,AC,XMDES,PRSUB,SFBEP)
      CALL CDADDI(AT,AC,XMO,PR,XMT,XMDES,PTO,TTO,CDAD,CDADS,ABYPAC,
     1 ABLEAC,AOAC,WA,SFBEP,SFBPP)
      CALL CDBYPA(XMO,ABYPAC,ABLEAC,CDBE,CDBP,PRTOT,PINF,PTO)
      CALL CDAUXI(AUAENG,CDAUX)
      CALL CDDIVI(XMO,AWAENG,CDDIV)
      IF (KEYZ.EQ.1) RETURN
   10 CALL CDBTA(XMO,XNPR,PTNOZ,TTNOZ,AENG,CDBT,WA,AC,ANOZ,BETA)
      CALL ENGCDI(XMO,EN,SDDG,CDI,FIP,Q,AC,XNPR)
      RATIO=AC*EN*PCDFAC/SWING
      CDADP=CDAD*RATIO*SFADP
      CDADSP=CDADS*RATIO*SFADSP
      CDBEP=CDBE*RATIO
      CDBPP=CDBP*RATIO
      CDAUXP=CDAUX*RATIO*SFAUXP
      CDDIVP=CDDIV*RATIO*SFDIVP
      CDBTP=CDBT*RATIO*SFBTP
      CDIP=CDI*RATIO*SFIP
      CDINLP=CDADP+CDADSP+CDBEP+CDBPP+CDAUXP+CDDIVP
      CDAFTP=CDBTP+CDIP
      CDINSP=(CDINLP+CDAFTP)*SFINSP
      RETURN
      END
```

```
      SUBROUTINE CDAUXI(AUAENG,CDAUX)                                   CDAU000
C                                                                       CDAU000
C     REF:  INTERNAL AERODYNAMICS MANUAL, NAR, PP 7-24 SEC 7.8          CDAU000
C     CDAUX = AUXILIARY SYSTEMS DRAG                                    CDAU000
C     AUAENG = AREA OF AUXILIARY SYSTEMS DIVIDED BY AC                  CDAU000
C                                                                       CDAU000
      CDAUX=2.*AUAENG                                                   CDAU000
      RETURN                                                            CDAU000
      END                                                               CDAU000
```

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OF POOR QUALITY

```
      SUBROUTINE PRINL(XM0,DELPR,PR,IPR,PRTOT,PRSUB,XMPRI,XPRI,PRDES,
     1 XMDES)
C
C     SUBROUTINE TO COMPUTE THE INLET TOTAL PRESSURE RECOVERY
C     IPR = PRESSURE RECOVERY BRANCH CODE
C         = 1, AIA STANDARD
C         = 2, MIL SPEC 5008B
C         = 3, NORMAL SHOCK
C         = 4, TABLE LOOK-UP, PR VS MACH
C     XM0  = FREE STREAM MACH NUMBER
C     DELPR = INCREMENTAL PRESSURE RECOVERY REDUCTION
C             INPUT AS A POSITIVE NUMBER
C     PRTOT = TOTAL PR TO INLET FACE
C
      DIMENSION XMPRI(6),XPRI(6)
      IPRDUM=0
      XM02=XM0*XM0
      XMD2=XMDES*XMDES
      IF (IPR.GE.0) GO TO 10
      IPRDUM=IPR
      IPR=IABS(IPR)
   10 IF (IPR.LE.10) GO TO 20
      IPRDUM=IPR
      IPR=IPR-10
   20 GO TO (30,40,50,70),IPR
C
C     AIA STANDARD
C
   30 PR=1.
      PRDES=1.
      IF (XM0.GT.1.) PR=1.-.1*(XM0-1.)**1.5
      IF (XMDES.GT.1.) PRDES=1.-.1*(XMDES-1.)**1.5
      GO TO 60
C
C     MIL SPEC 5008B
C
   40 PR=1.
      PRDES=1.
      IF (XM0.GT.1.) PR=1.-.075*(XM0-1.)**1.35
      IF (XMDES.GT.1.) PRDES=1.-.075*(XMDES-1.)**1.35
      GO TO 60
C
C     NORMAL SHOCK
C
   50 PR=1.
      PRDES=1.
      IF (XM0.GT.1.) PR=(6.*XM02/(XM02+5.))**3.5*(6./(7.*XM02-1.))**2.5
      IF (XMDES.GT.1.) PRDES=(6.*XMD2/(XMD2+5.))**3.5
     . *(6./(7.*XMD2-1.))**2.5
   60 PR=PR-DELPR
      PRDES=PRDES-DELPR
      GO TO 120
C
C     TABLE LOOK-UP
C
   70 PR=XPRI(1)
```

```
      SLP=(XPRI(6)-XPRI(5))/(XMPRI(6)-XMPRI(5))                          PRIN005
      IF (XMD.LT.XMPRI(1)) GO TO 90                                      PRIN005
      DO 80 I=1,5                                                        PRIN005
      XMTEST=XMPRI(I)                                                    PRIN005
      IF (XMD.GE.XMTEST.AND.XMD.LE.XMPRI(I+1)) PR=XPRI(I)+(XMD-XMPRI(I))PRIN006
     1 /(XMPRI(I+1)-XMTEST)*(XPRI(I+1)-XPRI(I))                          PRIN006
      IF (XMD.GE.XMTEST.AND.XMD.LE.XMPRI(I+1)) GO TO 90                  PRIN006
   80 CONTINUE                                                           PRIN006
      PR=XPRI(6)+SLP*(XMD-XMPRI(6))                                      PRIN006
      IF (PR.LT..1) PR=.1                                                PRIN006
   90 PRDES=XPRI(1)                                                      PRIN006
      IF (XMDES.LT.XMPRI(1)) GO TO 110                                   PRIN006
      DO 100 I=1,5                                                       PRIN006
      XMTEST=XMPRI(I)                                                    PRIN007
      IF (XMDES.GE.XMTEST.AND.XMDES.LE.XMPRI(I+1)) PRDES=XPRI(I)+(XMDES-PRIN007
     1 XMPRI(I))/(XMPRI(I+1)-XMTEST)*(XPRI(I+1)-XPRI(I))                 PRIN007
      IF (XMDES.GE.XMTEST.AND.XMDES.LE.XMPRI(I+1)) GO TO 110             PRIN007
  100 CONTINUE                                                           PRIN007
      PRDES=XPRI(6)+SLP*(XMDES-XMPRI(6))                                 PRIN007
      IF (PRDES.LT..1) PRDES=.1                                          PRIN007
  110 PRSUB=1.                                                           PRIN007
  120 PRTOT=PR*PRSUB                                                     PRIN007
      IF (IPRDUM.NE.0) IPR=IPRDUM                                        PRIN007
      RETURN                                                             PRIN008
      END                                                                PRIN008
```

REFERENCES

1. Gregory, T. J.: Computerized Preliminary Design at the Early Stages of Vehicle Definition. NASA TM X-62,303, 1973.

2. Ball, W. H.: Propulsion System Installation Corrections. USAF Contract AFFDL-TR-72-147, The Boeing Company, 1972.

3. Sibulkin, Merwin: Theoretical and Experimental Investigation of Additive Drag. NACA Report 1187, 1954.

4. Lighthill, J. J.: Higher Approximations in Aerodynamic Theory. Princeton University Press, 1960.

5. Little, F. K.: Internal Aerodynamics Manual. North American Rockwell Corp., Columbus, Ohio, 1968.

6. Aerospace Handbook. General Dynamics, Fort Worth, Nov. 1962.

7. Model Specification E1075-A, Engine, Aircraft, Turbojet. J-85-GE-13, General Electric Company, June 30, 1969.

8. Crane, C. A.; Spencer, B. L.; and Vance, C. H.: Flight Manual and Standard Aircraft Characteristics Performance Substantiation — F-5A/B Tactical Fighter with Two J-85-GE-13 Engines. NOR-67-29, Northrop Corporation, Norair Division, July 1967.

9. Kline, R. E.; Crews, A. H.; and Knight, W. J.: F-5A Category II Performance Test. Technical Report No. 65-15, Air Force Flight Test Center, 1965.

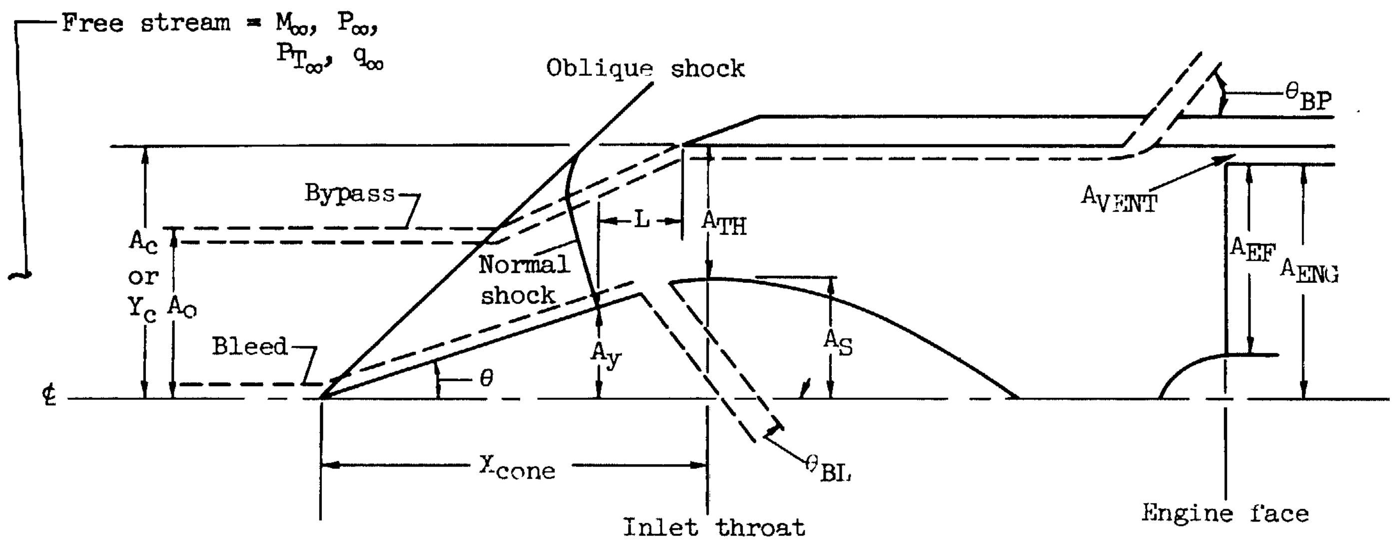
10. Hesse, W. J.; and Mumford, N. V. S.: Jet Propulsion for Aerospace Applications. Pitman Publishing Corp., 1964.

11. Ames Research Staff: Equations, Tables, and Charts for Compressible Flow. NACA Report 1135, 1953.

12. Barry, F. W.: Conical Flow Properties for Use in the Design of Supersonic Inlets, Report M-1266-1, Research Department, United Aircraft Corporation, Nov. 18, 1958.

13. Shapiro, A. H.: The Dynamics and Thermodynamics of Compressible Fluid Flow, vol. I. Ronald Press Co., 1953.

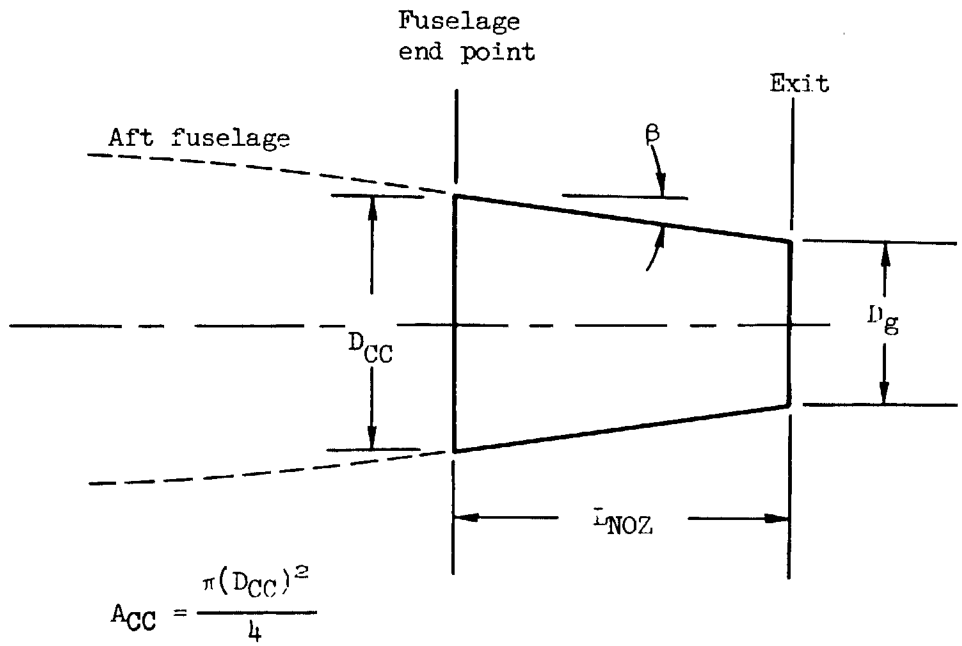
14. Anderson, W. E.; and Nelms, Walter P., Jr.: Performance and Control of Air Induction Systems for a Commercial Supersonic Transport — 1. Geometry and Steady State Performance. Presented at the AIAA/AFSC Vehicle Design and Propulsion Meeting, Nov. 1963.



Note: A denotes areas

(a) Inlet.

Figure 1.- Nomenclature.



(b) Nozzle.

Figure 1.- Concluded.

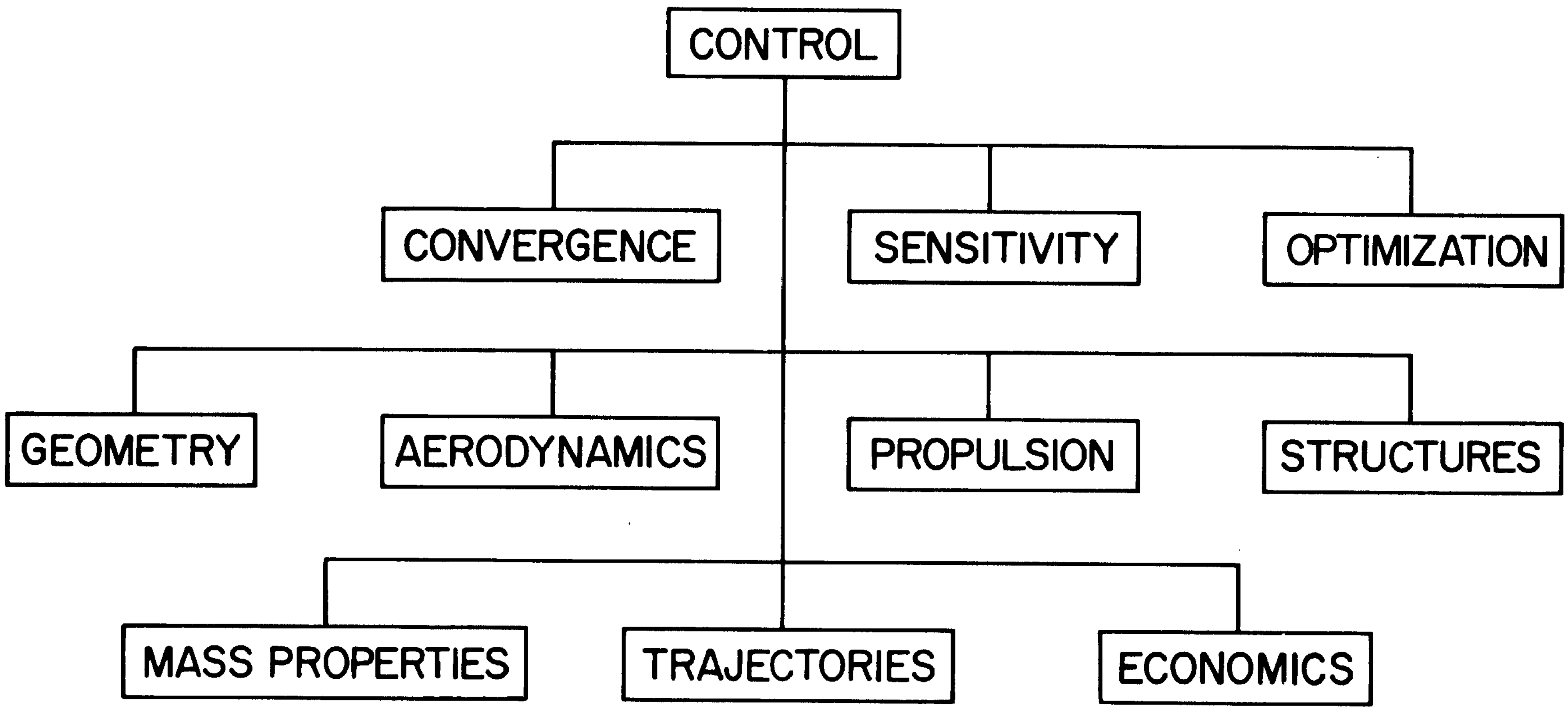


Figure 2.- Block diagram of ACSYNT.

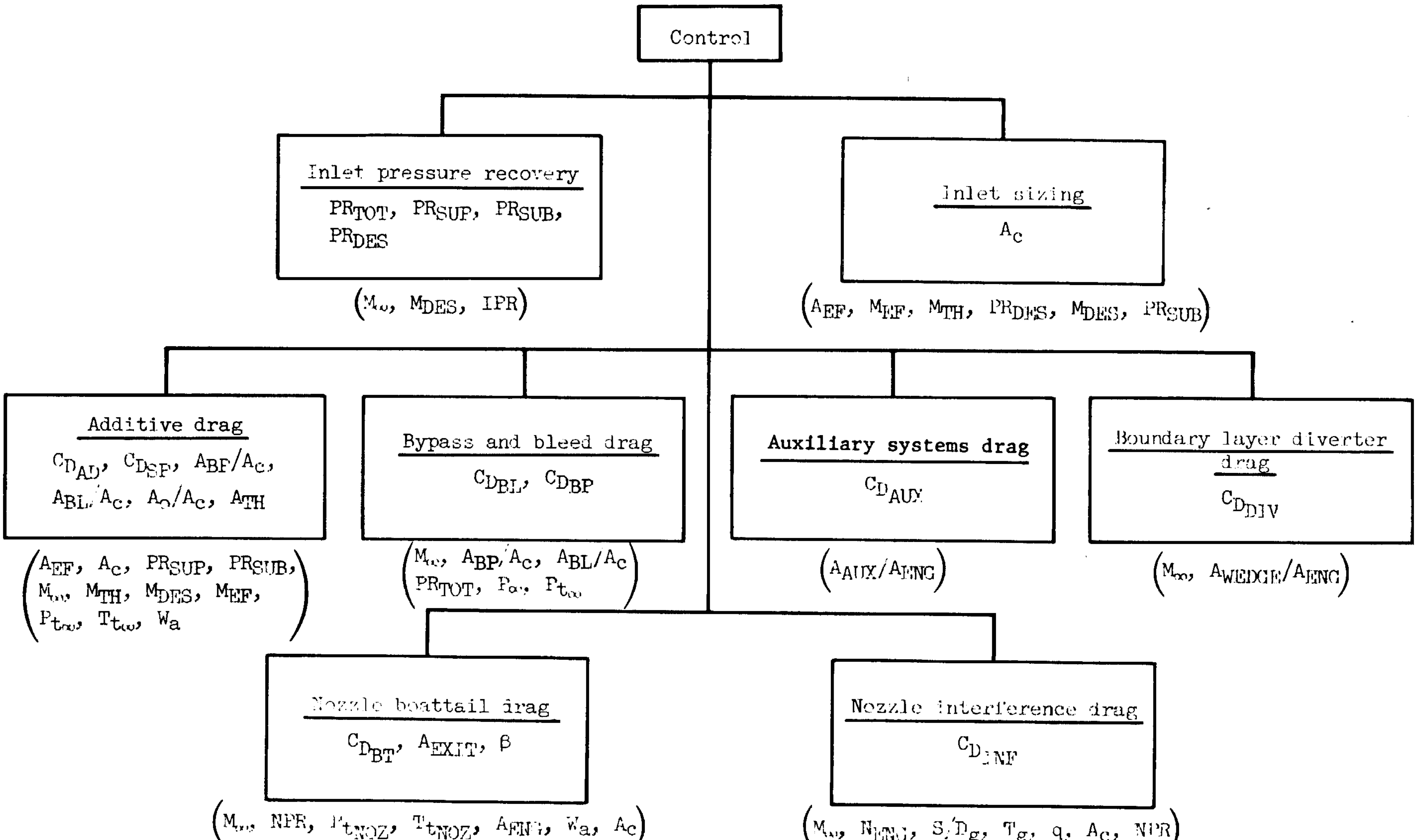
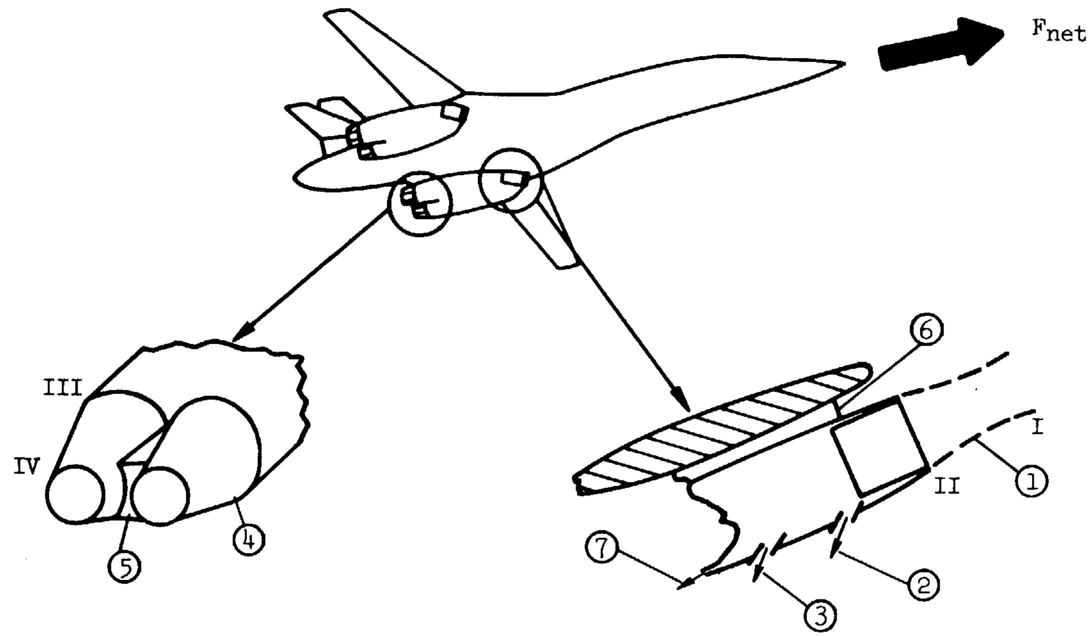


Figure 3.- Block diagram of propulsion installation losses subroutine (PRINC); values computed are in the boxes; inputs are in parentheses below each box.



Installation drag coefficients:

(1) Additive $= \int_{I}^{II} \frac{(P - P_\infty) dA_X}{q_\infty A_C}$

(2) Bleed $= \frac{\dot{m}_{BL} V_\infty - \left(\dot{m}_E V_E + A_E (P - P_\infty)\right)}{q_\infty A_C}$

(3) Bypass $= \frac{\dot{m}_{BP} V_\infty - \left(\dot{m}_E V_E + A_E (P - P_\infty)\right)}{q_\infty A_C}$

(4) Boattail $= \int_{III}^{IV} \frac{(P - P_\infty) dA_X}{q_\infty A_C}$

(5) Interference $= \int \frac{(P - P_\infty) dA_X}{q_\infty A_C}$

(6) Diverter $= \int \frac{(P - P_\infty) dA_X}{q_\infty A_C}$

(7) **Auxiliary** $= \frac{\dot{m}_{AUX} V_\infty}{q_\infty A_C}$

Accounting method:

$F_{net} = T_{INST} - D$

T_{INST} = installed thrust

D = airframe drag

Where:

$T_{INST} = T_{UNINST} - \left((1) + (2) + (3) + (4) + (5) + (6) + (7)\right) \times q_\infty A_C$

Where:

T_{UNINST} = uninstalled thrust (corrected for pressure recovery)

Figure 4.- Accounting method used in PRINC module.

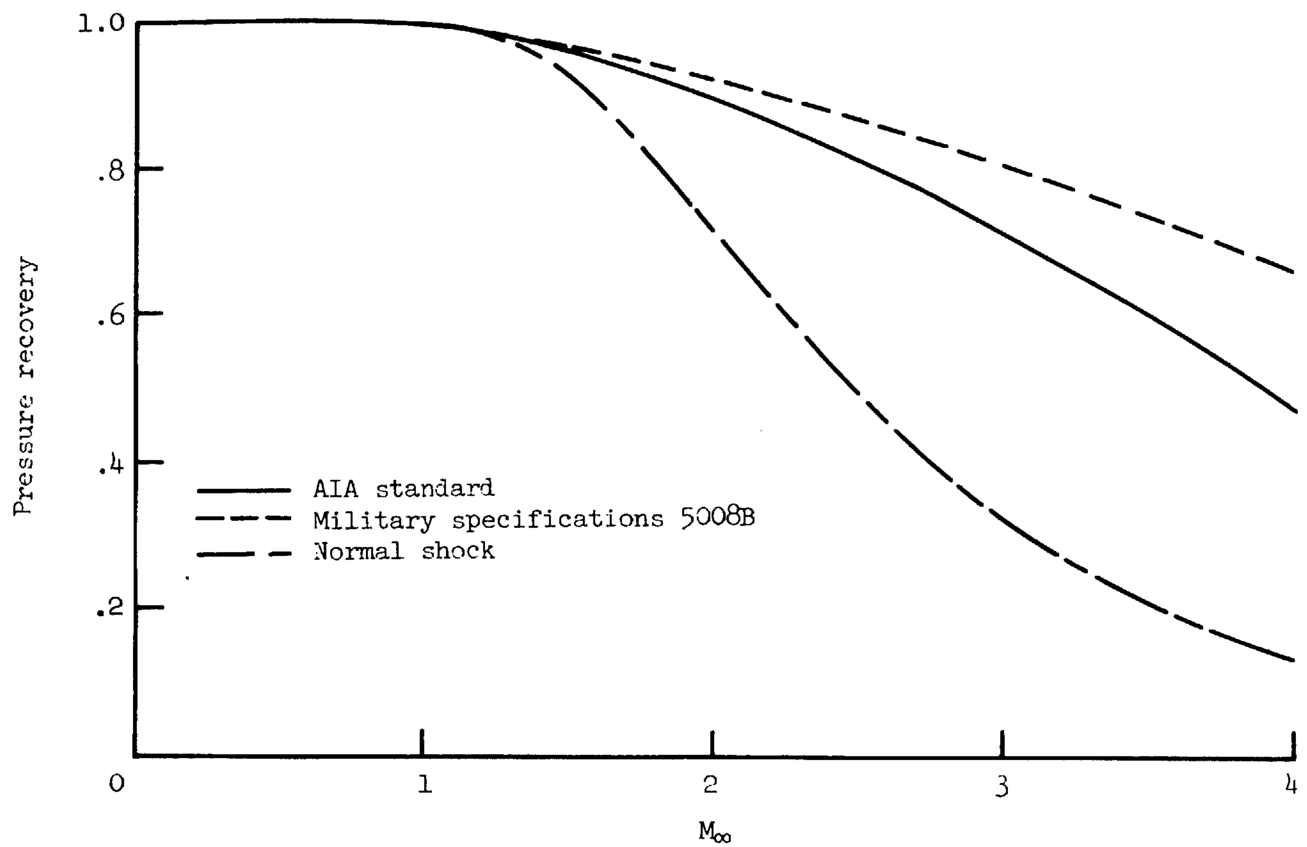


Figure 5.- Supersonic diffuser pressure recovery schedules.

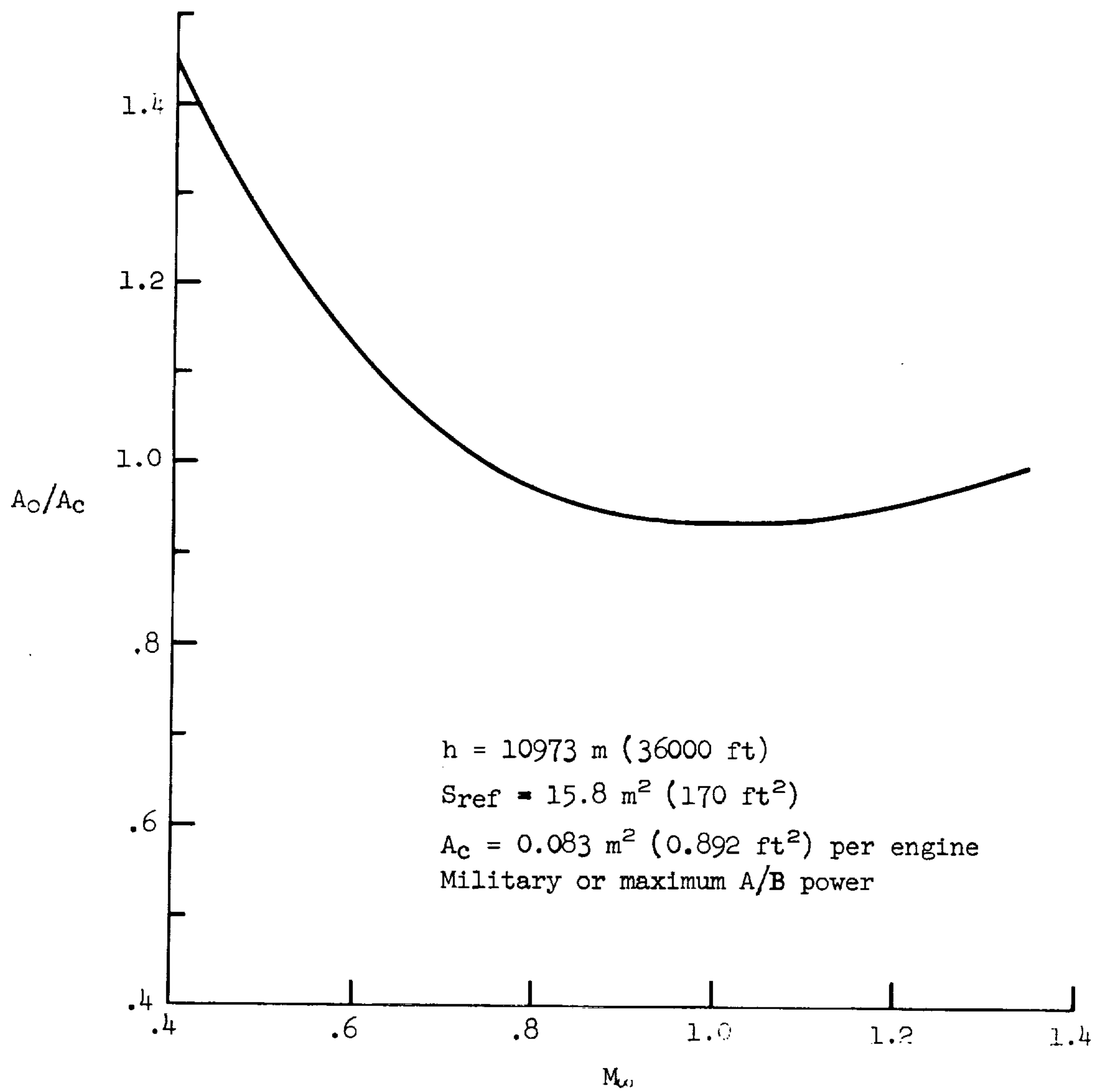
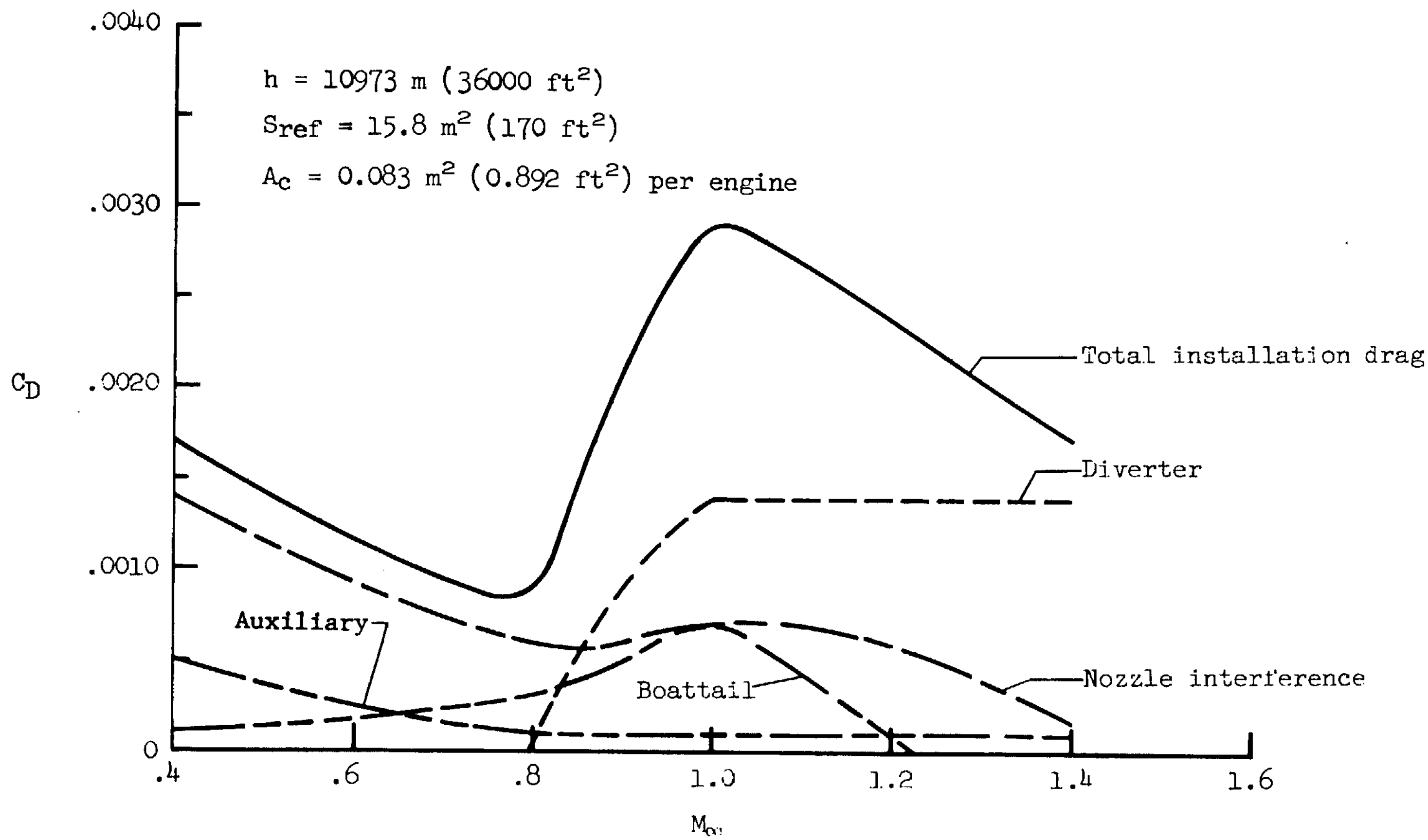
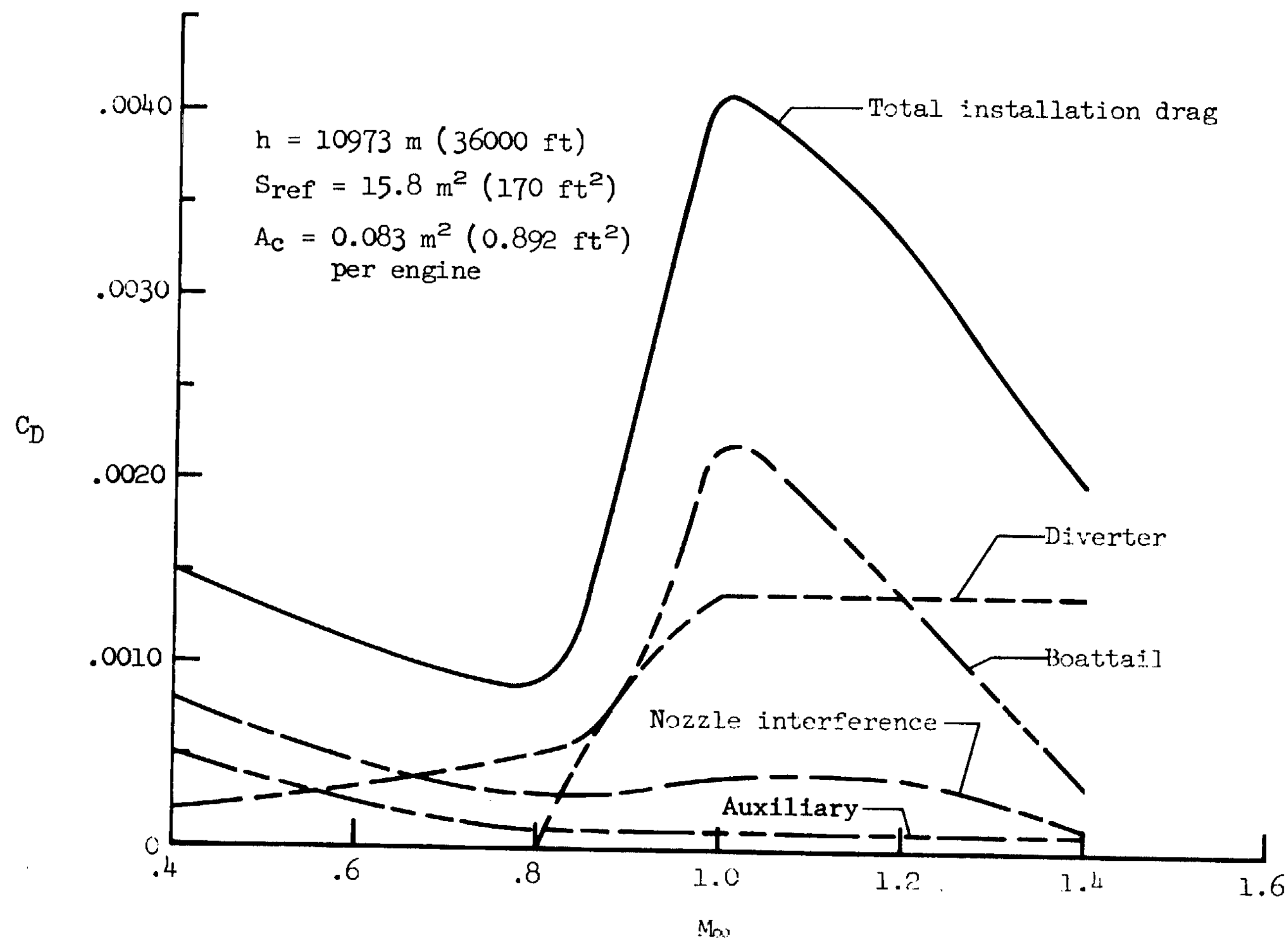


Figure 6.- Mass flow ratio versus Mach number for simulated F-5A with (2) J85-13 engines.



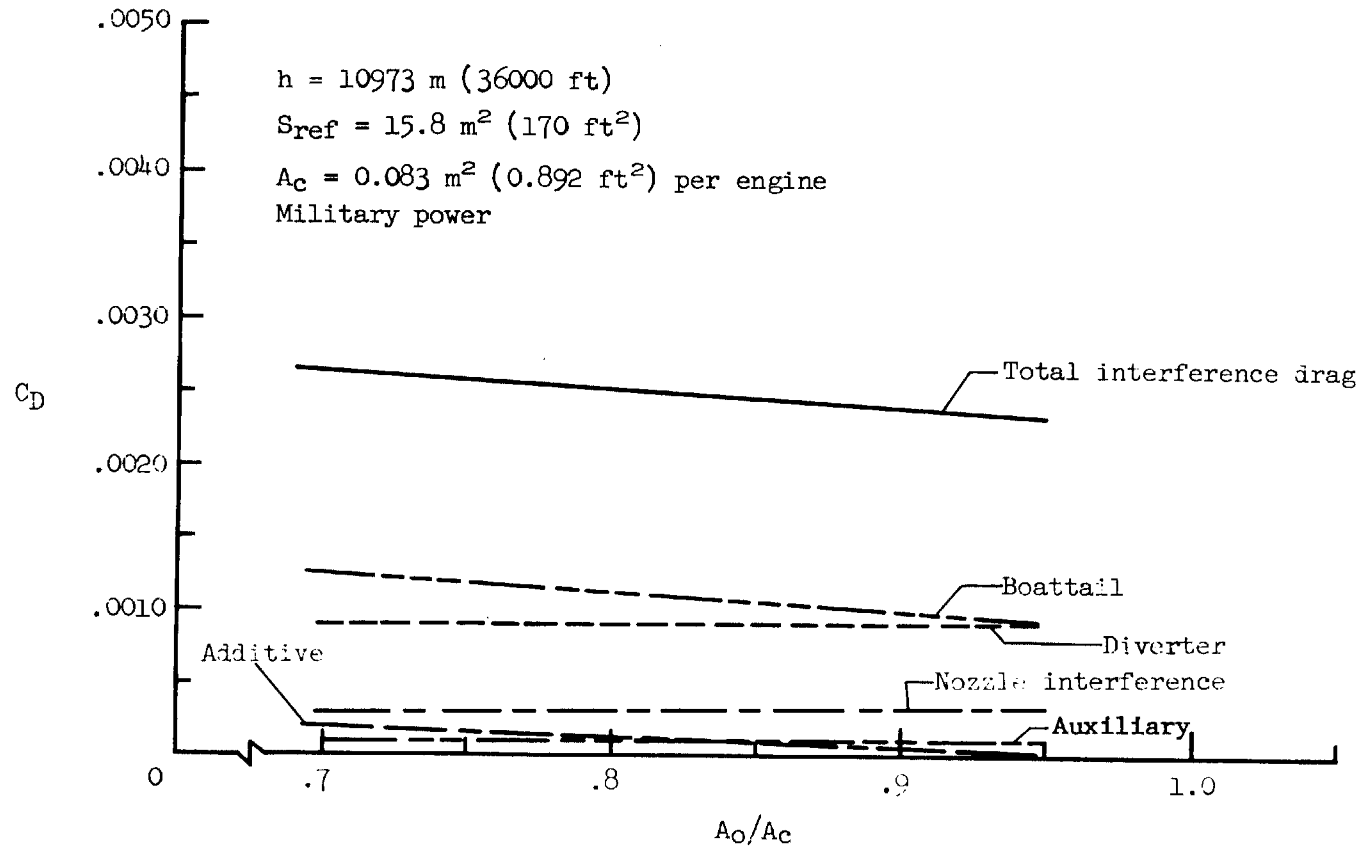
(a) Maximum A/B power setting.

Figure 7.- Example propulsion installation drag calculated by PRINC for simulated F-5A with (2) J85-13 engines; based on S_{ref}.



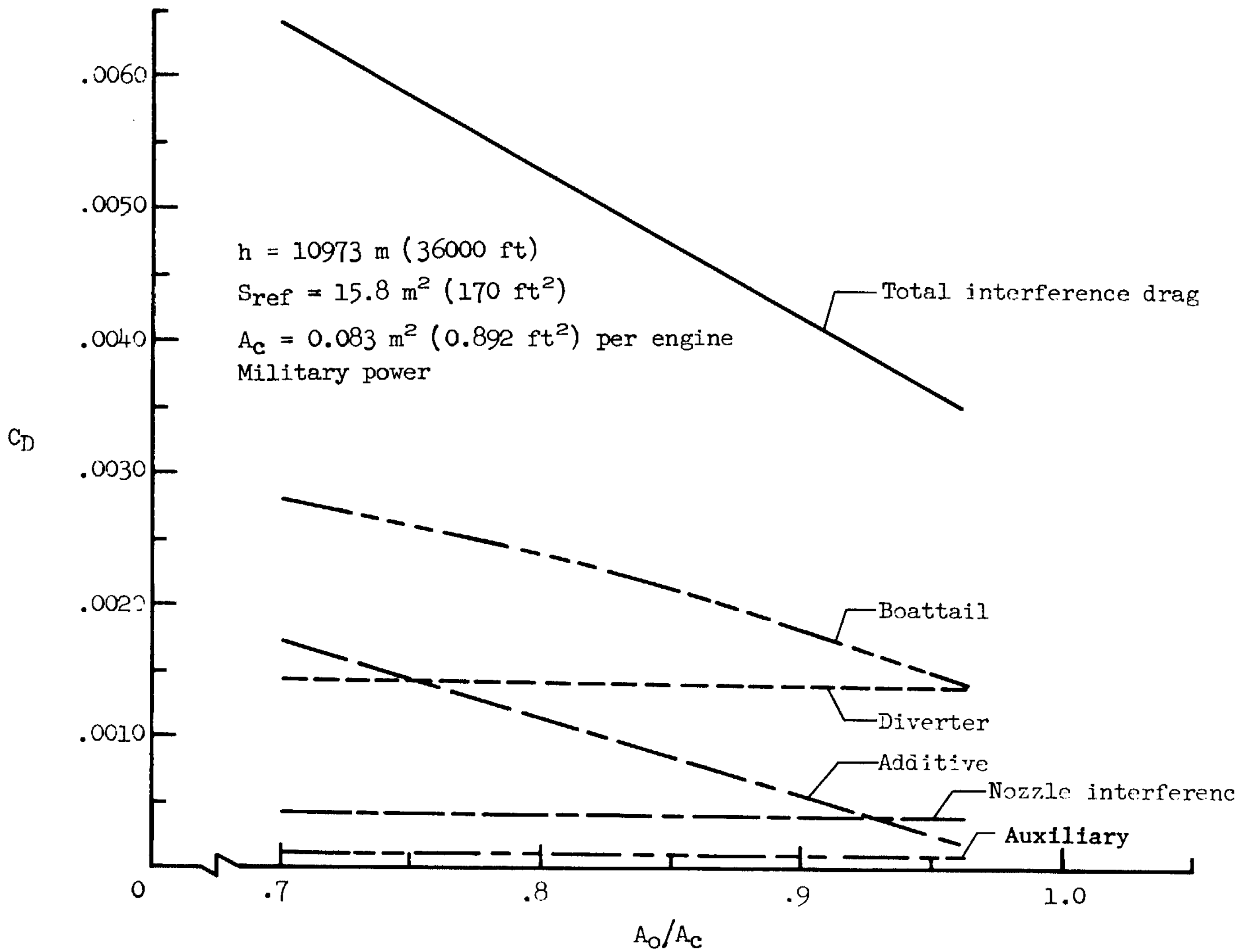
(b) Military power setting.

Figure 7.- Concluded.



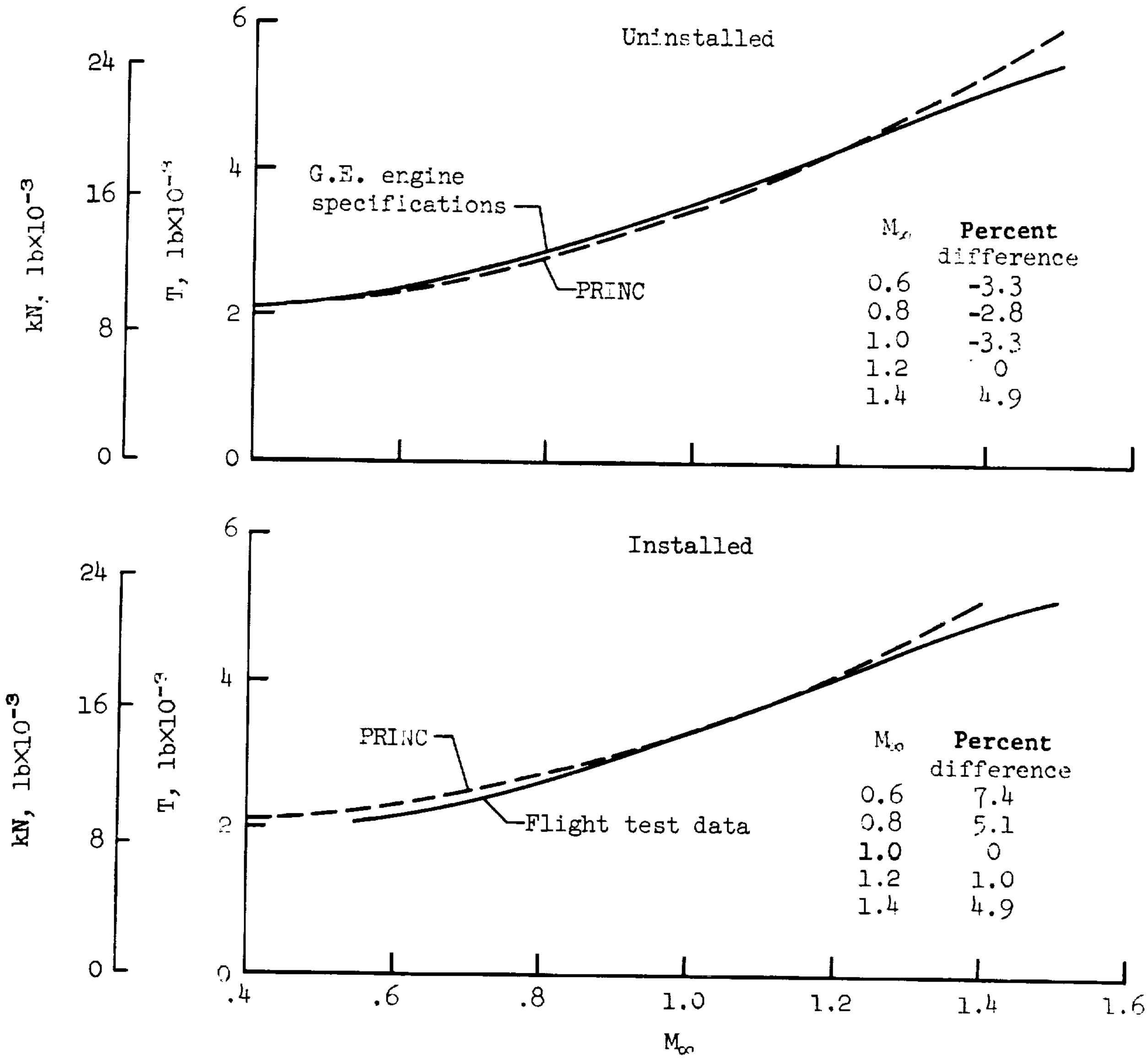
(a) $M_\infty = 0.9$

Figure 8.- Example installation drag versus mass flow ratio calculated by PRINC for simulated F-5A with (2) J85-13 engines; based on S_{ref}.



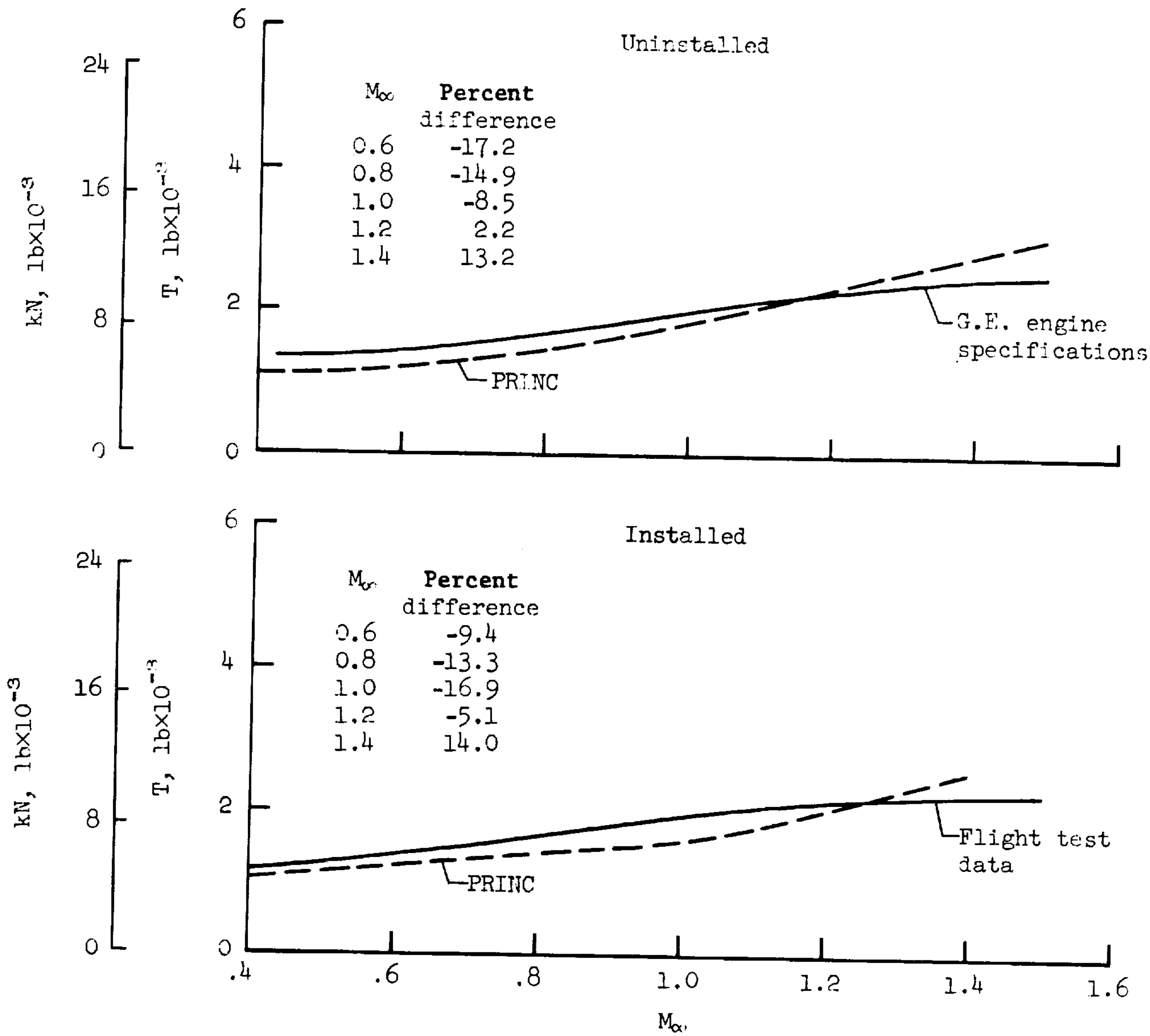
(b) $M_\infty = 1.2$

Figure 8.- Concluded.



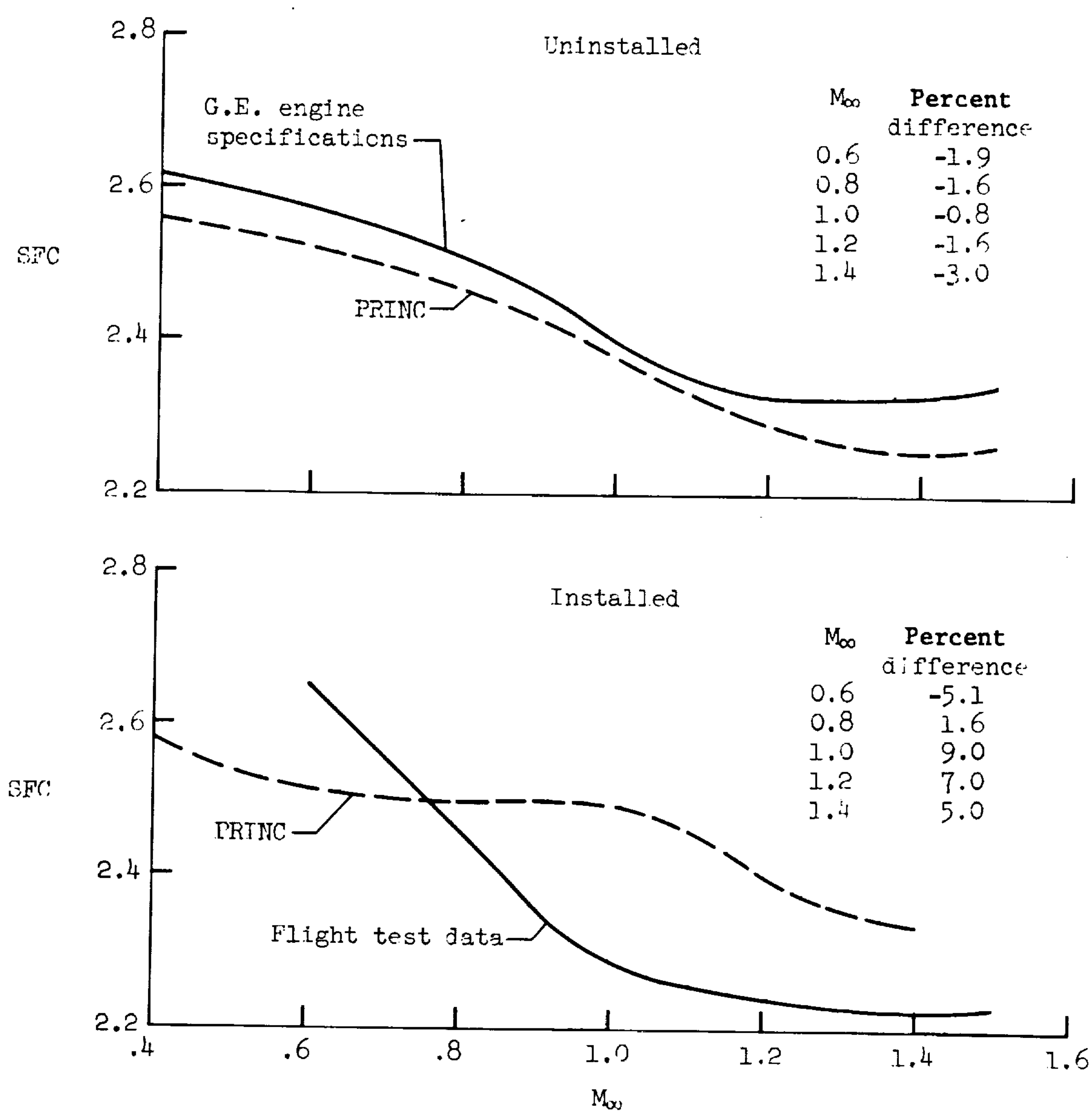
(a) Maximum afterburning.

Figure 9.- Thrust correlation for simulated F-5A with (2) J85-13 engines; h = 10973 m (36000 ft).



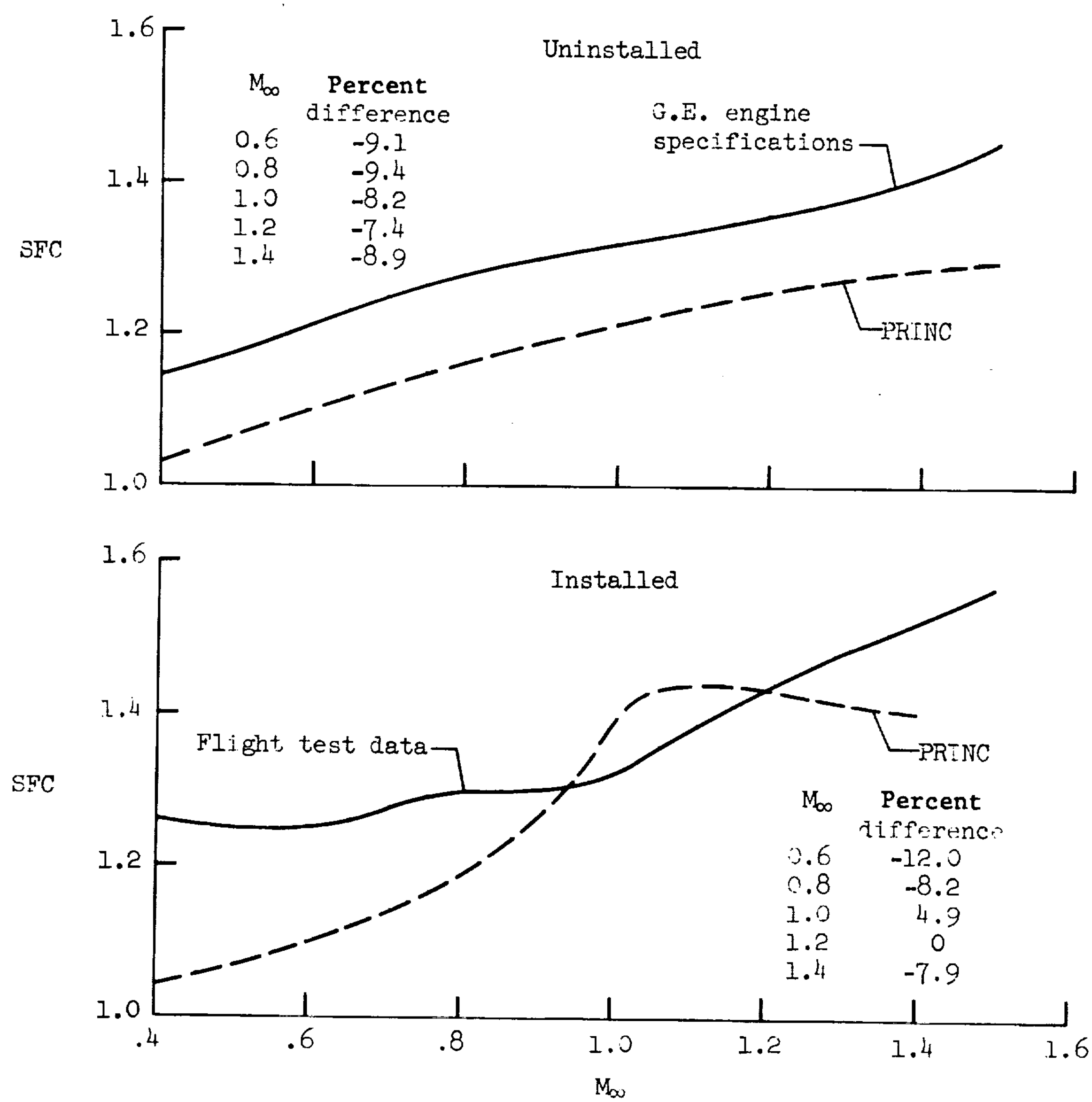
(b) Military power.

Figure 9.- Concluded.



(a) Maximum afterburning.

Figure 10.- Specific fuel consumption correlation for simulated F-5A with (2) J85-13 engines; h = 10973 m (36000 ft).



(b) Military power.

Figure 10.- Concluded.

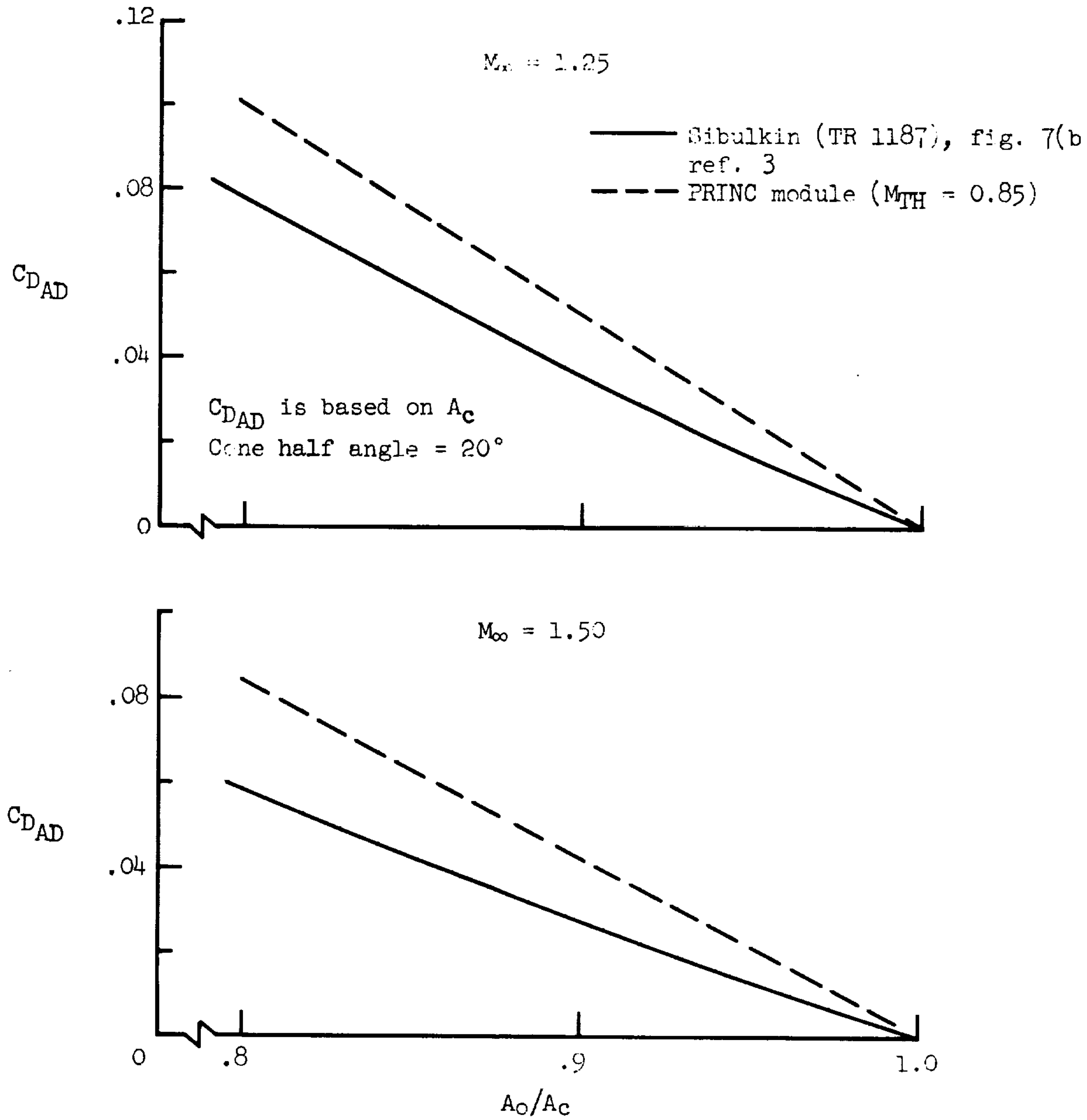


Figure 11.- Additive drag correlations.

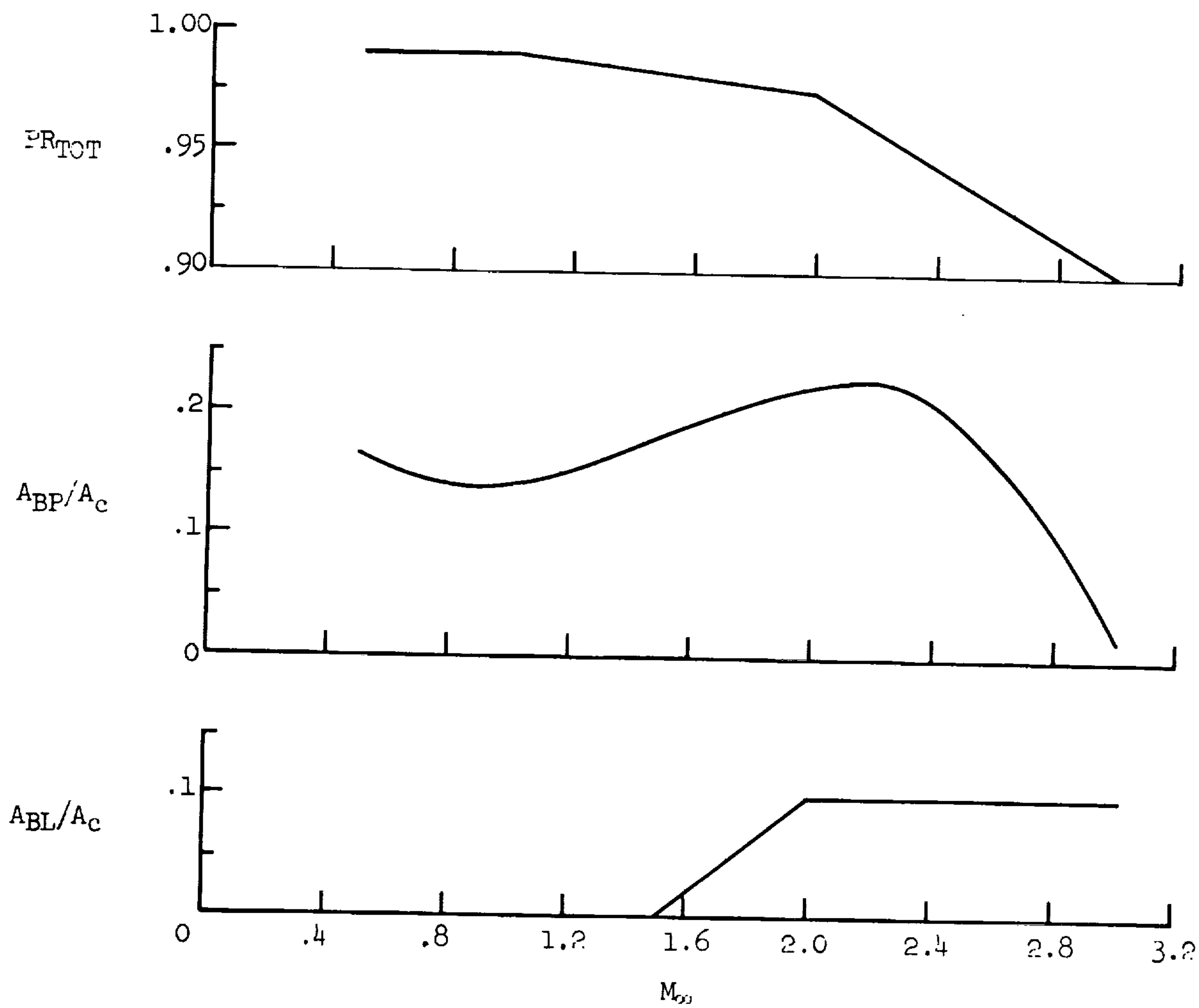
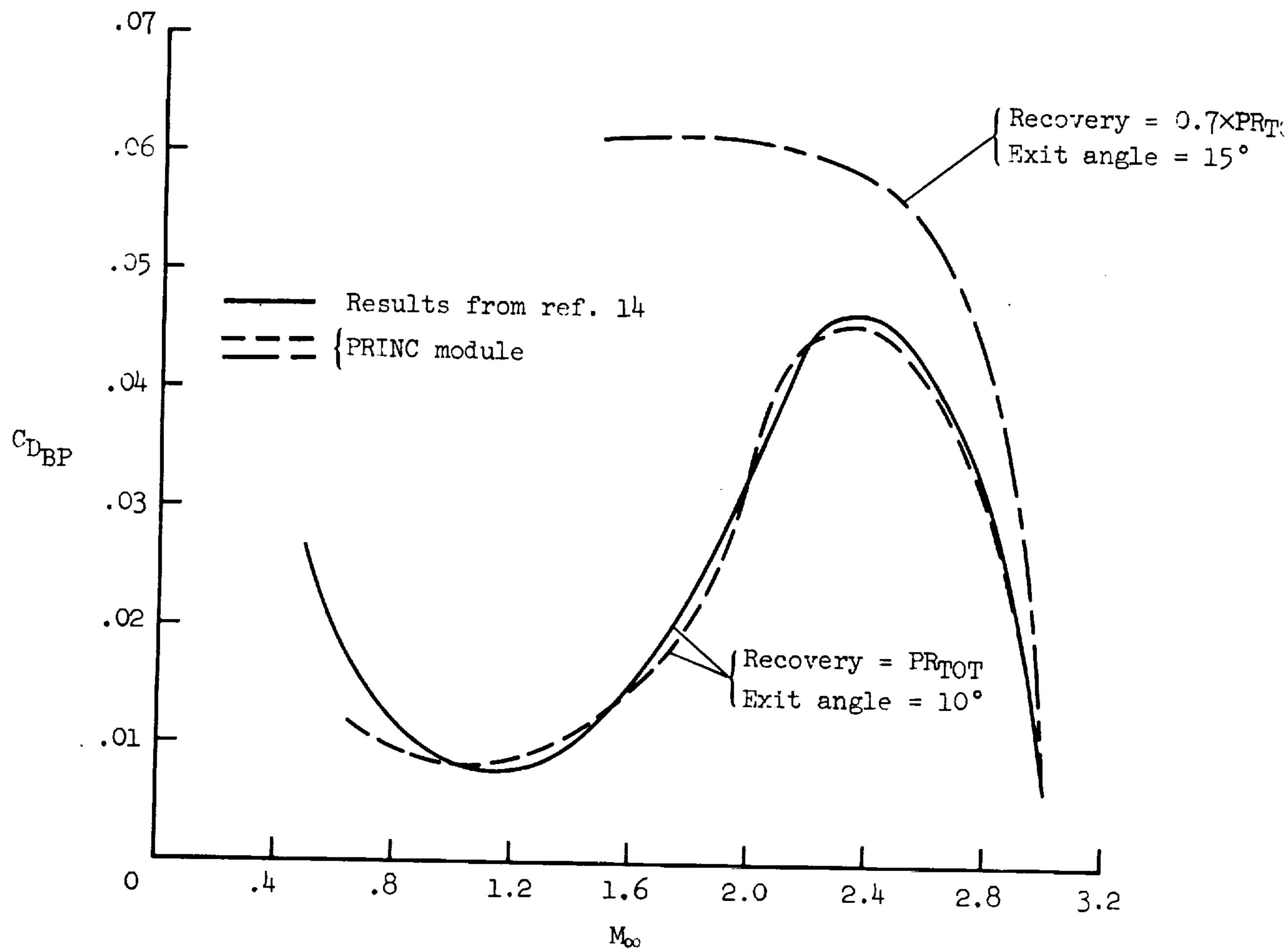
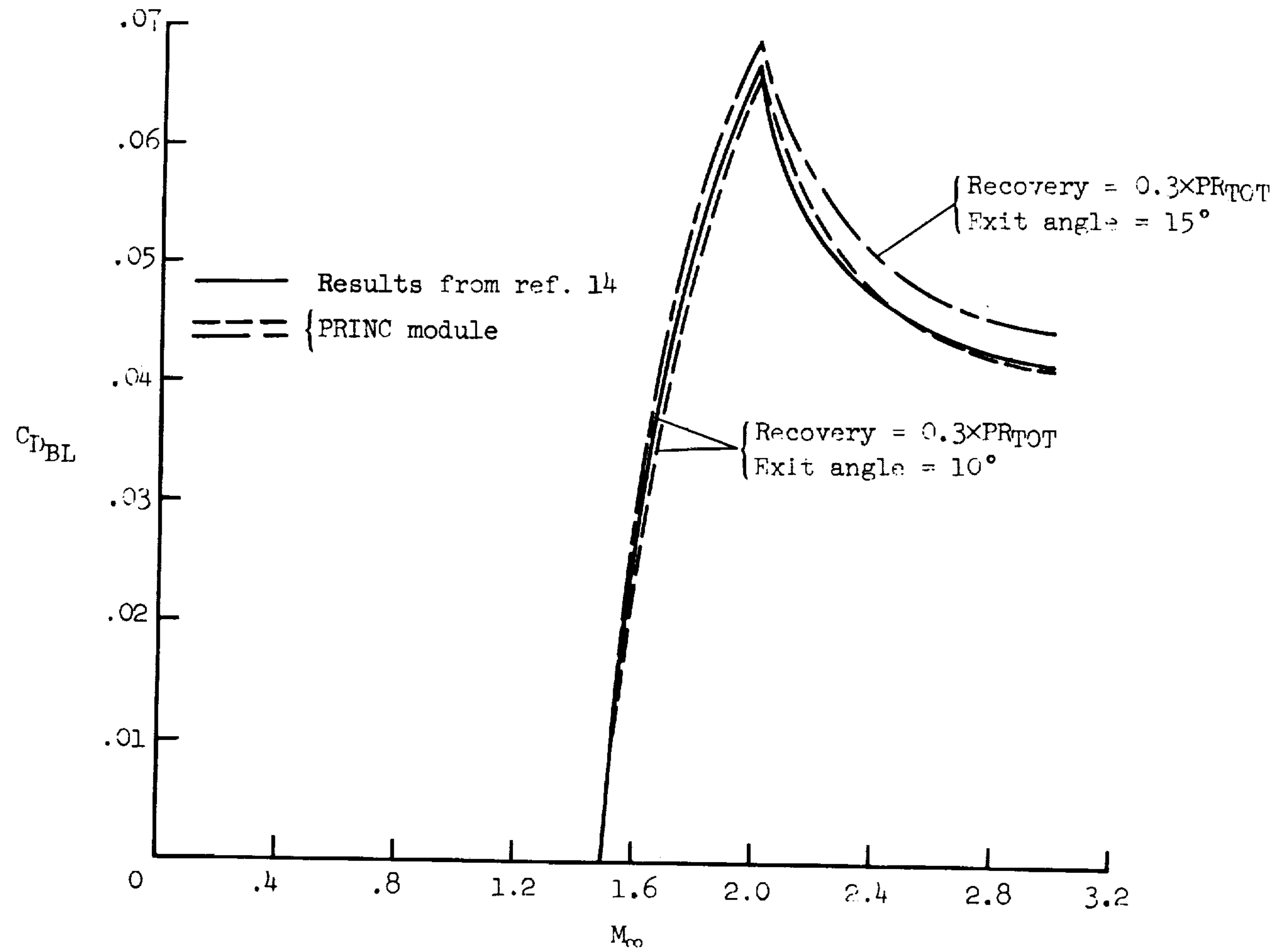


Figure 12.- Pressure recovery and mass flow schedules for a study supersonic transport concept from reference 14.



(a) Bypass drag.

Figure 13.- Correlation of bypass and bleed drag coefficients for sonic exit Mach numbers; based on capture area.



(b) Bleed drag.

Figure 13.- Concluded.

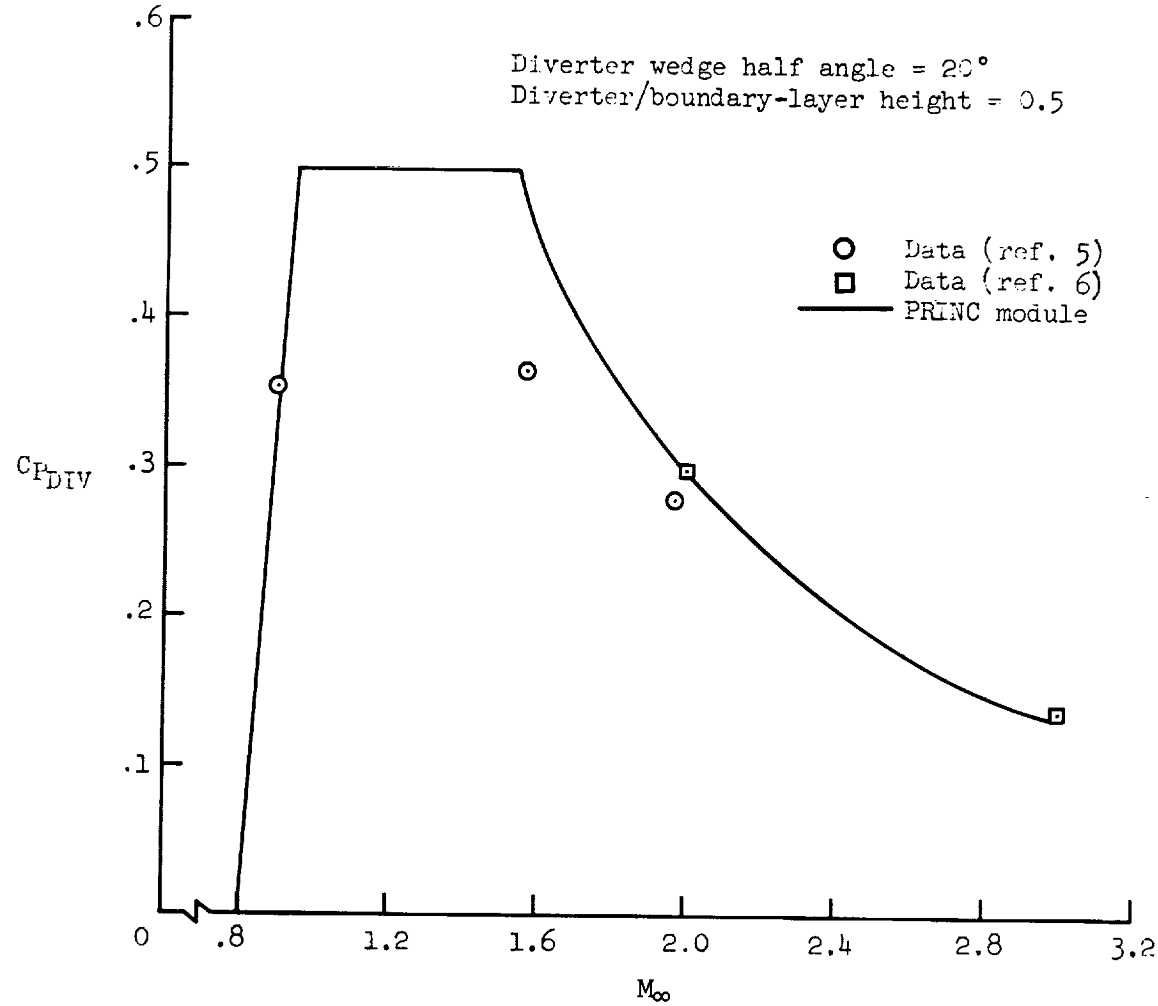
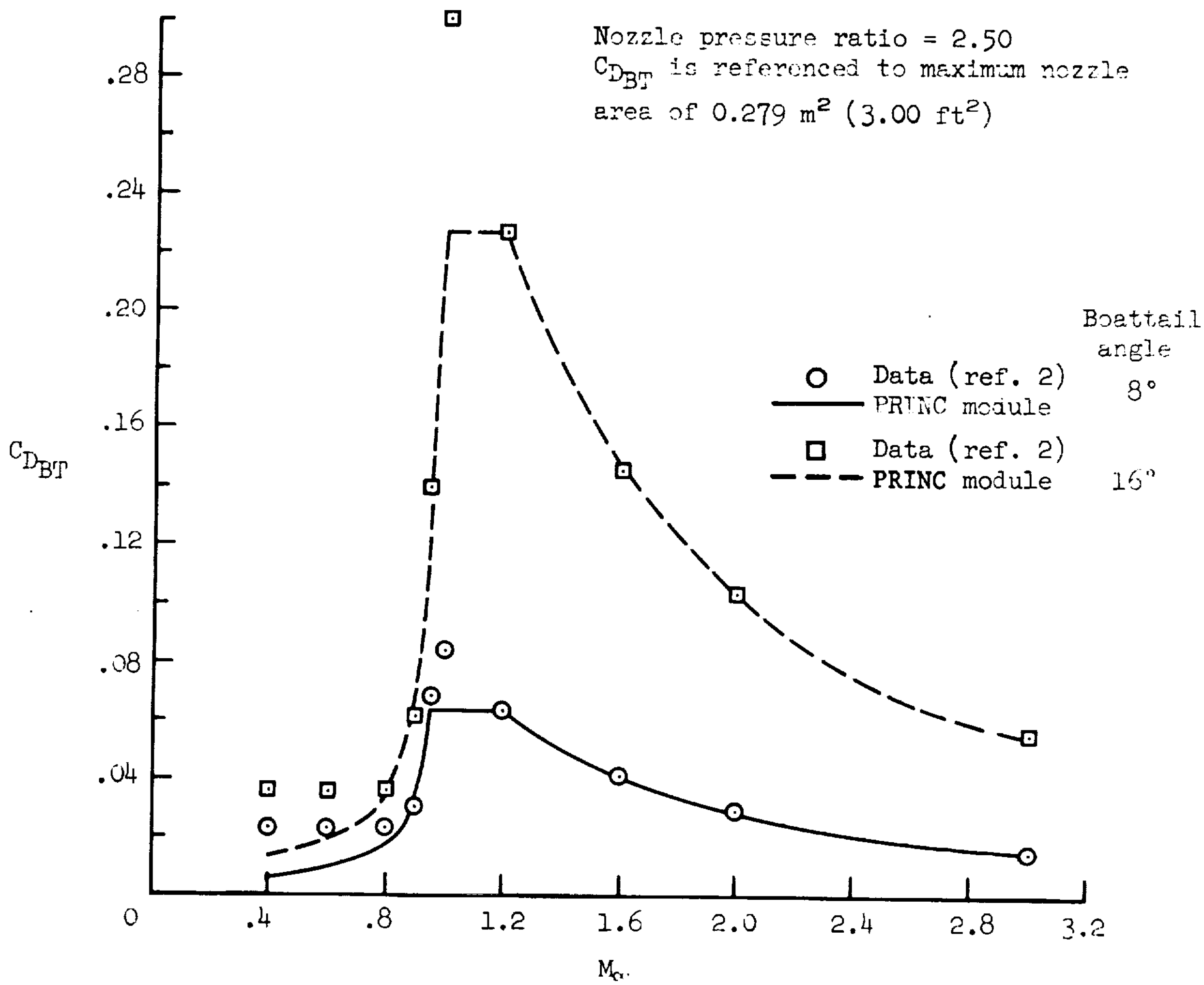
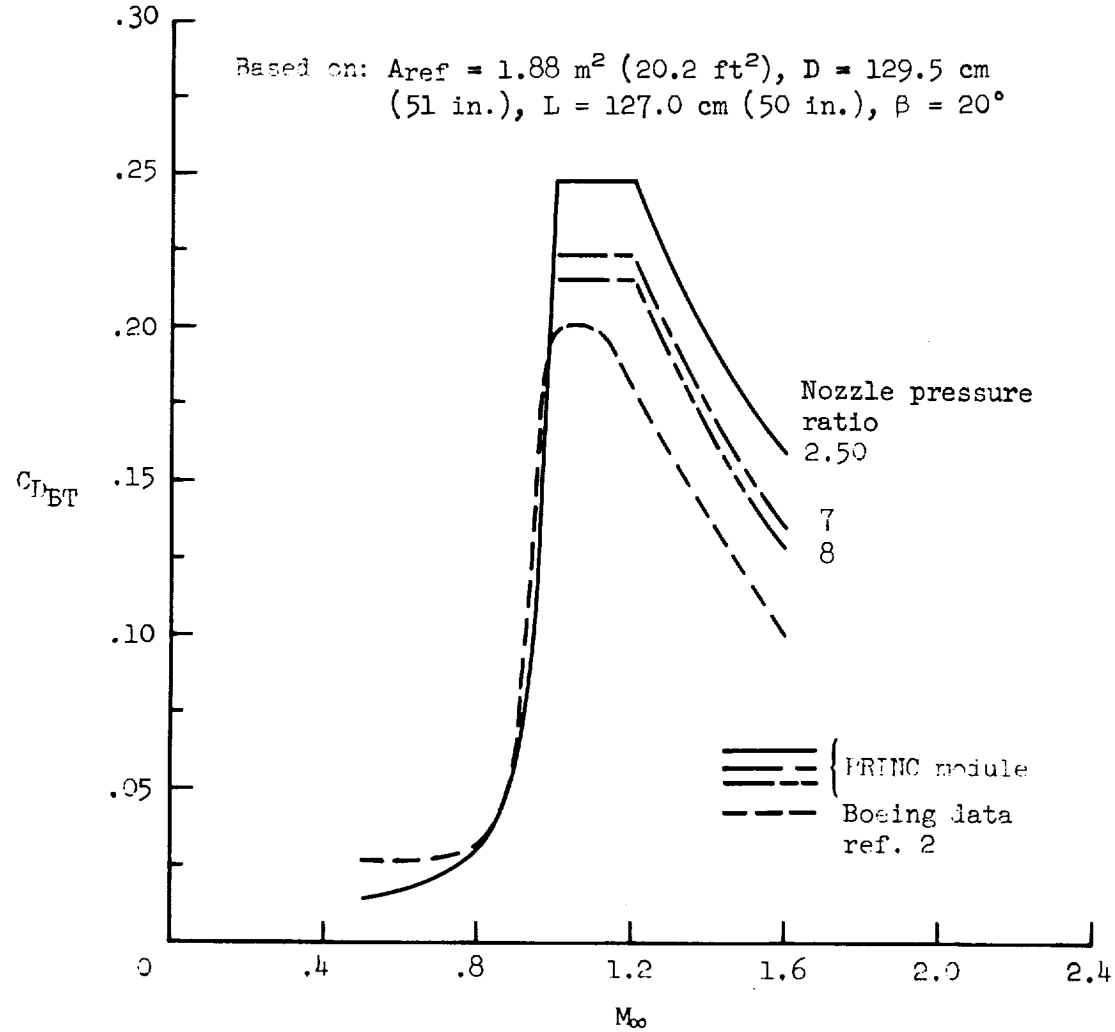


Figure 14.- Correlation of boundary-layer diverter pressure coefficient.



(a) Boattail angle effects.

Figure 15.- Correlation of nozzle boattail drag coefficient.



(b) Nozzle pressure ratio effects.

Figure 15.- Concluded.

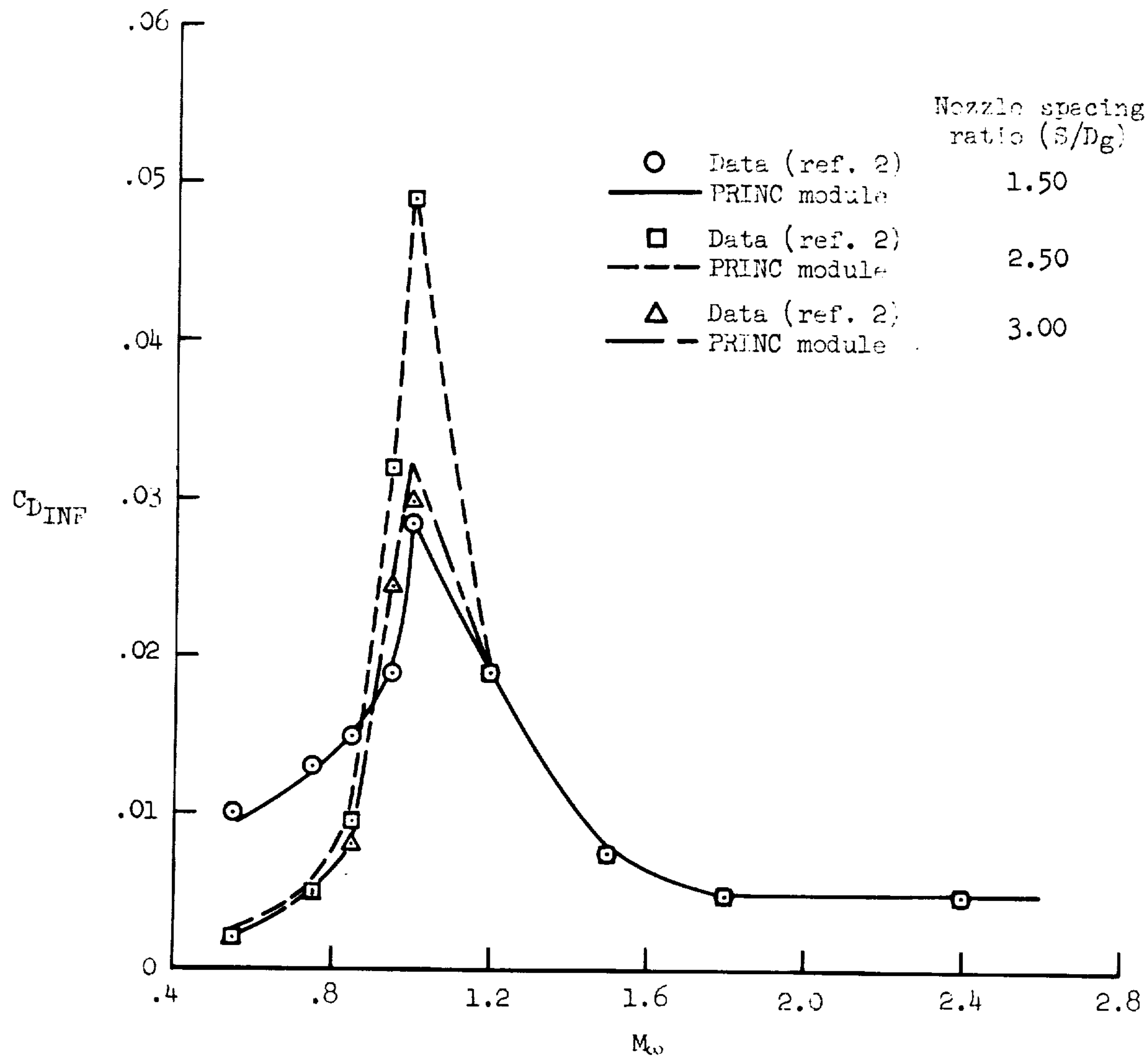


Figure 16.- Correlation of nozzle interference drag coefficient; nozzle pressure ratio = 2.50.